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Structural requirements and launcher validation process for MECSE CubeSat (versão corrigida após defesa)

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To my grandfather,
Manuel Francisco José.

"The noblest pleasure is the joy of understanding"

Leonardo da Vinci

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Resumo

MECSE é o primeiro CubeSat desenvolvido na UBI - Universidade da Beira Interior e é um nanosatélite em desenvolvimento resultante de uma parceria entre o C-MAST - Center for Mechanical and Aerospace Science and Technologies e o CEiiA - Centre of Engineering and Product Development. O objectivo do MECSE, além de ser educacional e de providenciar experiência prática a alunos universitários em projetos espaciais, tem como missão demonstrar que é possível manipular a camada de plasma, usando um campo electromagnético que irá permitir a mitigação da perda de sinal de rádio frequência que ocorre quando um veículo espacial reentra na atmosfera terrestre.

Nesta dissertação, uma visão geral dos requisitos para a configuração, design, dimensionamento, verificação e validação são apresentados usando diversas referências, sendo dos documentos da ECSS - European Cooperation for Space Standardization que a maior parte da informação foi consultada, de forma a identificar e apresentar os requisitos de uma perspetiva da engenharia de sistemas e estrutural.

Posto isto, foi inicialmente identificado os principais requisitos estruturais, tais como o ambiente mecânico, a interconexão entre CubeSat e lançador e a frequência natural mínima exigida à estrutura do satélite. De seguida foram assinaladas as condições pelas quais as verificações e validações se devem realizar numa estrutura espacial. Tendo as condições de verificação e validação levado à definição dos métodos de verificação e à organização, planeamento e metodologia dos processos de verificação que normalmente são aplicados num CubeSat para a sua validação. Sabendo que a validação só é obtida se forem seguidas as condições definidas para a realização das verificações numéricas e experimentais, tal como das ROD's - Reviws of Design e das inspeções a proceder.

Numa fase final deste trabalho, foi analisado um conjunto de lançadores com o objetivo de identificar uma proposta adequada para o projeto MECSE, tendo sido os lançadores Bloostar, Electron, LauncherOne e o Vector-R com melhor desempenho para os parâmetros analisados. A análise dos vários lançadores foi realizada também com o intuito de reconhecer qual o ambiente mecânico mais exigente de entre os casos tidos em conta, de forma a que o projeto MECSE possa ser desenhado e analisado segundo esse mesmo caso, enquanto o lançador final não é selecionado. Também nesta fase é realizada uma proposta para uma possível abordagem ao processo de verificação, com o principal foco para os modelos numéricos a desenvolver, para a metodologia de testes experimentais, foi definida uma abordagem híbrida com o intuito de ser construido um modelo estrutural, um modelo de qualificação de engenharia e um modelo protoflight, tal como é definido os níveis e duração dos testes a realizar nesses mesmos modelos numéricos e experimentais.

Palavras-chave

Nanosatélite; CubeSat; MECSE; ECSS; Requisitos estruturais; Verificação estrutural; Verificação numérica; Verificação experimental; Ambiente Mecânico; Lançadores de satélites.

Abstract

MECSE is the first CubeSat being developed at UBI - University da Beira Interior, and it is an under development nanosatellite, resulting from the collaboration between C-MAST - Center for Mechanical and Aerospace Science and Technologies and CEiiA - Centre of Engineering and Product Development. MECSE's mission, aside from the education aims to provide hands-on experience to universitary students in space projects, it intends to demonstrate that is possible the manipulation of plasma layer using an electromagnetic field that will mitigate the RF - Radio Frequency blackout, which occurs when a space vehicle re-enter in the Earth's Atmosphere.

In this dissertation, an overview of the requirements for a structural configuration, design, dimensioning, verification and validation are presented, using several references. Nevertheless, the ECSS - European Cooperation for Space Standardization documents was where the most of the information was consulted, in order to identify and present the requirements from a systems engineering and structural perspective.

Therefore, it was initially identified the main structural requirements, such as the mechanical environment, the interconnection between CubeSat and launcher, and the minimum natural frequency required for the satellite structure. Followed by the main structural requirements are the conditions under which the verifications and validations must be performed in a satellite structure. This led to the definition of the verification methods and to the organization, planning and methodology of the verification processes, which are normally used for a CubeSat validation. Knowing that the validation is only granted if the verifications and validation conditions are followed, applied and accomplished in the numerical and experimental verifications, such as for ROD's - Reviews of Design and inspections.

In a final phase of this work, a set of launchers was analysed with the objective of identifying a suitable proposal for MECSE project. The launchers Bloostar, Electron, LauncherOne and Vector-R were the launchers with better performance for the analysed parameters. The analysis of the various launchers was also carried out in order to recognize the most demanding mechanical environment among the cases taken into account, so that MECSE project could be designed and analysed according to the worst case scenario, while the final launcher is not selected. In this same phase a proposal is made for a possible approach to the verification process, with the main focus on the numerical models to be developed, on the experimental test methodology which was defined by a hybrid approach with a structural model, an engineering qualification model and a protoflight model, as well as identified the levels and duration of the tests and analyses to be performed in these same numerical and experimental models.

Keywords

Nanosatellite; CubeSat; MECSE; ECSS; Structural requirements; Structural verification; Numerical verification; Experimental verification; Mechanical Environment; Launch Vehicles.

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Abbreviated Terms

ACS	Attitude Control Subsystem
ADC	Attitude Determination and Control subsystem
AIT	Assembly, Integration and Test
AITP	Assembly, Integration and Test Plan or just Test Plan
AIVP	Assembly, Integration and Verification Plan
AL	Acceptance Loads
AR	Acceptance Review
CAD	Computer Aided Design
Cal Poly	California Polytechnic State University
CDR	Critical Design Review
CLA	Coupled Load Analysis
CRR	Commissioning Result Review
CT	ECSS Categorie
C&DH	Command and Data Handling
C-MAST	Center for Mechanical and Aerospace Science and Technologies
DBL	Design Buckling Load
DBP	Design Burst Pressure
DCA	Departamento de Ciências Aeroespaciais
DDF	Design Definition File
DJF	Design Justification File
DLL	Design Limit Loads
DUL	Design Ultimate Loads
DYL	Design Yield Loads
ECSS	European Cooperation for Space Standardization
ELR	End of Fife Review
EQM	Engineering Qualification Model
EM	Engineering Model
ESA	European Space Agency
FA	Fatigue Analysis
FEA	Finite Element Analysis
FEM	Finite Element Model
FM	Flight Model
FRR	Flight Readiness Review
FOS	Factor of Safety
FOSY	Yield design factor of safety
FOSU	Ultimate design factor of safety
GEO	Geosynchronous Orbit
GSE	Ground Support Equipment
GPS	Global Positioning System

IandT	Integration and Test
ICD	Interface Control Document
ILS	Integrated Logistic Support
ISECG	International Space Exploration Coordinate Group
ISO	International Organization for Standardization
ISS	International Space Station
j proof	Proof factor
j burst	Burst factor
KA	Acceptance factor
Kld	Local design factor
Km	Model factor
Kp	Project factor
KQ	Qualification factor
LL	Limit Load
LEO	Low Earth Orbit
LRR	Launch Readiness Review
MCC	Mission Control Center
MCR	Mission Control-out Review
MDRA	Modal and Dynamic Response Analysis
MMDD	Mathematical Model Description and Delivery
MPD	Maximum Design Pressure
MDR	Mission Definition Review
MECSE	Magnetohydrodynamics/Electrohydrodynamics CubeSat Experiment
MoS	Margin of Safety
NASA	National Aeronautics and Space Administration
NDI	Non Destructive Inspections
NSR	Northern Sky Researcher
N/A	Not Applicable
OBS	Organizational Breakdown Structure
ORR	Operational Readiness Review
PA	Product Assurance
PAP	Product Assurance Plan
PBS	Project Breakdown Structure
PCB	Printed Circuit Board
PDR	Preliminary Design Review
PFM	Protoflight Model
PMP	Project Management Plan
POCC	Payload Operation Control Center

P&ODC	Position and Orbit Determination and Control
PP	Proof Pressure
PRD	Project Requirements Documents
PropS	Propulsion Subsystem
PRR	Preliminary Requirements Review
PSD	Power Spectrum Density
PTR	Post Test Review
P-POD	Poly Picosatellite Orbital Deployer
QL	Qualification Loads
QM	Qualification Model
QR	Qualification Review
QSL	Quasi-Static Loads
RF	Radio Frequency
ROD	Review Of Design
S&MS	Structures and Mechanics Subsystem
SEP	System Engineering Plan
SOCC	Spacecraft Operations Control Center
SOW	Statement Of Work
SPL	Sound Pressure Level
SRR	System Requirements Review
SRS	Shock Response Spectrum
SSA	Stress and Strength Analysis
SSDL	Stanford University's Space Systems Development Laboratory
SM	Structural Model
STM	Structural-Thermal Model
TAC	Tests-Analysis Correlation
TCS	Thermal-Control Subsystem
TPRO	Test Procedures
TRB	Test Review Board
TRL	Technology Readiness Level
TRR	Test Readiness Review
TS	Technical requirements Specification
TP	Test Predictions
TSPE	Test Specifications
TT&C	Telemetry, Tracking and Command
U	Standardized cube with 10 cm side
UBI	Universidade da Beira Interior
URSS	Union of Soviet Socialist Republics
USA	United States of America
VCB	Verification Control Board
VCD	Verification Control Document
VP	Verification Plan

WBS	Work Breakdown Structure
WCA	Worst Case Analysis
WCS	Worst Case Scenario
ZEE	Zona Económica Exclusiva

Nomenclature

[K]	Stiffness Matrix
u	Grid Point Displacements
F	Applied Load Vector
[M]	Mass matrix
[B]	Damping matrix
A and B	Integraion Constants
ω_n^2	Circular Natural Frequency
f_n	Natural Frequency
M_r	Rigid body motion mass matrix
Φ_R	Rigid body motion of the Finite Element model
E_r	Rigid body motion strain energy
F_r	Rigid body motion residual nodal forces
δF	Residual force vector
δW	Residual work
ε	Work ratio
x, y, z	Axes directions
R x,y,z	Rotation in direction x, y z
x_{cog}, y_{cog}, z_{cog}	Centre of gravity in direction x, y, z
$I_{i,j}$	Moments of Inertia, being $i, j = x, y, z$

Chapter 1

Introduction

This chapter 1 begins with the reference of the National, European and World context of the project under analysis and with an indication of the reasons and ambitions of performing this work.

The second section Small Satellites presents an introduction to space systems and to the first satellites, alongside with a reference to a couple short-term projects in Portugal. To establish important concepts, a brief presentation on Small Satellites and CubeSats is also provided in conjunction with a concise description of Space Missions and Orbits, as well as the Spacecraft Configurations and Subsystems.

A decisive part of this work the MESCE - Magnetohydrodynamics/Electrohydrodynamics CubeSat Experiment is carefully discussed in this chapter for a correct understanding of all relevant points of the project.

1.1 Motivation

"The space is a place to be" - Robert Cleave.

"Across the sea of space, the stars are other suns" - Carl Sagan.

The exploration of the unknown, the desire for adventure and accomplishment, the anxiety to realize if we are the only ones in the Universe are intrinsic characteristics of human beings. Understand and solve these questions are scientific community goals [1].

The beginning of space exploration was triggered by these ambitions, but also by the desire of the greatest economies after the Second World War of showing their power, influence and scientific technology superiority in comparison with the rest of the world [2]. The first space explorations aimed to understand the space environment and to achieved it, the moon was explored, and the knowledge about other Solar System elements and our planet was improved. Inherently to these events are the scientific developments and advances of the use of x-rays, ultraviolet light, infrared or microwaves adopted in a vast number of missions [3].

These new explorations allowed innovation to take place and forced a change of mentality. Afterwards, this development sustained a huge improvement in the quality of life of everyone, with the integration of new technologies on a world wide scale. The introduction of television, navigation by GPS - Global Positioning System telecommunications, among other technologies used in the daily lives of humanity are examples of this [4].

Moreover, since the major developments of Micro and Nano Technology, the space research has gained new contours, seeing an exponential growth in spacecraft's launch [5, 6]. This relatively new technology allows the construction of smaller and cheaper components. The electronics development were the main contribution to the space growth nevertheless, also other areas lead to an improvement of space exploration. Rocket propulsion using nano-propellants or the possibility of a space elevator are other two applications of the nano technology [7, 8].

Connecting the reasons stated above with the short period of time to design and develop a Small satellite, and in particular a CubeSat, these conditions lead to the cost contraction in a spacecraft launch into orbit [9]. Currently, CubeSat's category of satellites is capable of executing what years ago just a Small satellite could meet. This transformation of the space market grants the opportunity of developing satellites not only for military or national organizations with a large economic expression as it used to be but also allows organizations or even companies that pretend to improve their chances in the space business. In the past, a small satellite over 40 kg was tagged with a price around 130 million euros, right now a CubeSat initial project required budgets in the neighbourhood of 250 thousand euros in agreement with Ref. [9]).

Open Cosmos Ltd is an example of an organization that aims to improve its opportunities in the space business, claiming to offer simple and affordable space access, arguing that are able to "*it can take its costumers' payloads from development to launch in under 10 months for just around 550 thousand euros. What under normal circumstances costs from 2.5 to 5 million euros*" [10] or the GAUSS Srl which has very distinct goals compared with the Open Cosmos Ltd. GAUSS Srl is an Italian company based in Rome, founded in 2012 as a spin-off of the Scuola di Ingegneria Aerospaziale of Sapienza with the aim of research, develop and implementing aerospace projects, with the added value of educational aspects and the execution of related cultural initiatives [11,12].

If it is true that the Portuguese expression in space scientific community is reduced, in comparison with space dominant countries such as USA - United States of America or Russia, it is also true that the Portuguese space science is far from being irrelevant to the scientific community.

Apart from the companies that develop every year projects with space application; aside from the participation in the development of new technologies, since the Portuguese access to ESA - European Space Agency almost two decades ago; the renewed approach of the Portuguese government for space specifies a new strategy for the space program in Portugal (identified in Diário da República, at 12 of March of 2018 [13]). Portugal sees not only the opportunity to promote the economic development, the creation of qualified employment, the generation of conditions for the development of the space industry in Portugal as a possibility but also looks with optimism at the monitorization of the ZEE - Zona Económica Exclusiva through a constellation of satellites. This would increase the number of favourable opportunities for space projects in all the segments of the space system, allowing a complete turn in the Portuguese aerospace paradigm [13, 14].

This new strategy intended to be applied until 2030 and will make the most of high-level educational institutions like UBI - Universidade da Beira Interior, which already has experience with space programs, such as the participation of UBI in the first Portuguese satellite, PoSat-I; the development of investigations in orbital dynamics and control of space systems or even the researches in materials with spacial applications are advantages for this new cycle. It will also add opportunities to companies concerned about their financial viability, but do not leave behind the creation of knowledge and the added value to society following a responsible and sustained way. Most important, this new strategy intents to collect and manage data on the different geographic points of Portugal, in order to promote the development and growth of the country [14].

When the future of space exploration is mentioned, it is indispensable to acknowledge the Global Exploration Roadmap. The interest of 14 space agencies in participating on the ISECG - International Space Exploration Coordinate Group to discuss the guidelines for expanding human and robotic presence in the Solar System is huge, which lead to a new edition of the Global Exploration reflecting the international effort in the preparation of space exploration missions with the objective of being a reference in the generation of new ideas and solutions addressed to the challenges of transport payloads to Mars orbit and surface. The Moon is used as a track to achieve the ultimate goal of Mars [15].

Bearing all that in mind, this dissertation intends to provide an optimized but generic description of the structural requirements and the validation process of a 3U CubeSat structure. In order to organize a set of steps that, if performed accordingly allows the acceptance for launch of a small satellite Mechanical Structure or a satellite with similar characteristics permitting the application of this project in future spacecraft.

The ultimate goal of this work is to make a case study in MECSE. Leading the necessary structural validation of a project that intends to make a significant contribution to the scientific community with the MHD/EHD - MagnetoHydroDynamics/ElecteoHydroDynamics (explained in more detail in section 1.3 - Mission) which if successfully implemented it will help the development of a spacecraft that intent to re-enter the earth atmosphere [16].

1.2 Small Satellites

1.2.1 Space System

In order to fully understand a spacecraft[1], it has to be recognized that an artificial satellite is never an independent unit since it is always part of a space system. In general, every space system includes three components: the space segment, which is the physical spacecraft; the launch segment, that transports the spacecraft to the desired orbit, using a launch vehicle; and the ground segment, usually built around the ground station, which can be subdivided into several sections and whose purpose is to collect, process and distribute the data produced by the spacecraft, as well as other functionalities like control and command of the spacecraft. A representation of these three segments is presented in Figure 1.1 [17, 18].

1.2.2 Space Missions

Each spacecraft is designed to perform a specific mission. Since the beginning of the Space era, the most common purposes for satellites are the communications, whether it is for commercial, governmental or even military. Nevertheless, there are other types of spacecraft and satellite missions, such as scientific observation missions, weather monitoring, surveillance or navigation [19, 20]. Table 1.1 shows a summary of applications and space activities currently performed.

[1] A spacecraft by definition is a vehicle or a vessel designed to operate beyond the von Kármán ellipsoid. In the space community, this term has two meanings, it is used to refer to a whole vehicle or just a platform which the payload is mounted on [17].

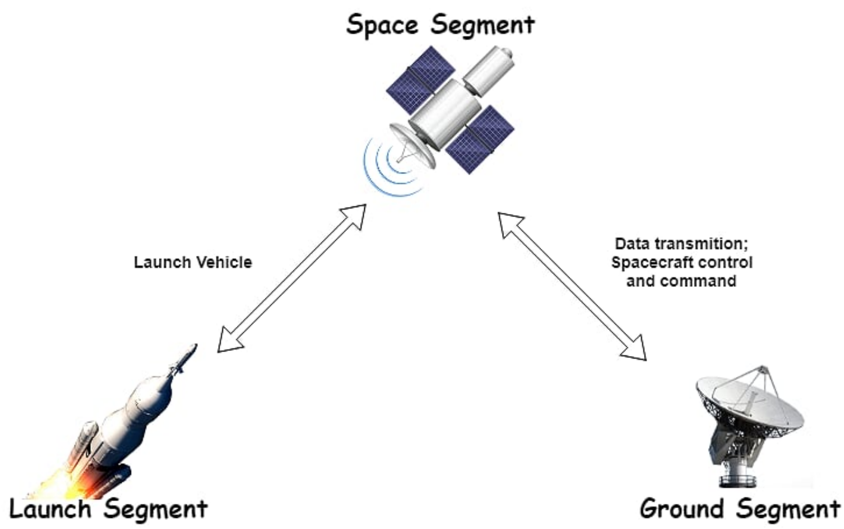

Figure 1.1: Sapce System, divided in the Launch Segment, Space Segment and Ground Segment [17].

Just like any other spacecraft, a CubeSat can have a wide variety of missions. Although their initial purpose was to train students and engineers, they have slowly become a big target for private companies with diverse goals, including commercial objectives. This is evident in Figure 1.2, which shows an overview of nanosatellites and CubeSats missions by the organization type from 1998 until 2018. Appendix A - Nanosatellites and CubeSat Database also presents the CubeSats mission but from other data source and in another time line [21–23].

Table 1.1: Categorization of Space Activities according to Ref. [17].

Category	Applications
Civil	Operational meteorology Search and rescue Technology demonstration Education and Training
Commercial	Satellite Communications and Broadcasting Global Navigation Satellite System Earth Observation Space Tourism Microgravity
Science	Astronomy and Space Science Earth Science
Military	Military Satellite Communications Global Navigation Satellite System Military Surveillance Satellite: Weather Satellites Military Surveillance Satellite: Imaging Satellites Military Surveillance Satellite: Electronic Surveilance Missile Defence Near Earth Objects

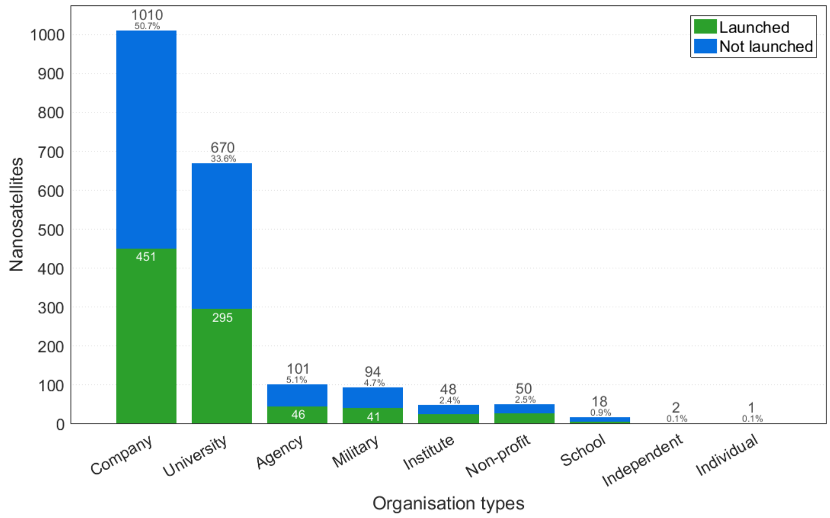

Figure 1.2: Nanosatellites by organisation type (data from Ref. [24]).

1.2.3 Small Satellites History

A small satellite by definition is a satellite with a mass under 500 kg [21]. Within the small satellites there are more categories, as shown in Table 1.2 .

Table 1.2: Classification of satellites by their mass (in agreement with Ref. [21]).

Class	Mass range [kg]
Large satellite	> 1000
Medium satellite	500 - 1000
Small satellite	< 500
Minisatellite	100 - 500
Microsatellite	10 - 100
Nanosatellite	1 - 10
Picosatellite	< 1

The first satellite to be launched into orbit was developed by the URSS - Union of Soviet Socialist Republics, now Russia, during the Cold War era. Sputnik I was a small satellite with a mass of 83 kg and a diameter of approximately 0.5 m. It was a great success, proving that a spacecraft could survive the hostile environment of space and, at the same time, it provides important information about the internal and skin temperature of an orbiting object [25, 26]. A few months later, the USA launched its first satellite, the Explorer I, a small satellite of 2 m long, with nearly 0.2 m of diameter, tipped by a blunt cone of 0.3 m and with a mass of 14 kg. This satellite's goal was to measure the radiation environment in orbit, and it marks the beginning of 75 satellites launched within the Explorer program [25, 27].

In Portugal, it was only in September 1993 that the first, and the only satellite was launched. PoSat-I was a microsatellite, operated until 2006, with the objective of providing secure communications and data transfer between military officers [28, 29]. In the near future, in 2020, Infante a microsatellite, will be the first satellite in orbit completely developed and built in Portugal. With an expected mass of 25 kg, it is the precursor of a constellation of twelve satellites which will support maritime surveillance and control of the proposed extension of the Portuguese continental shelf [30, 31].

Although micro and nanosatellites are much smaller and lighter compared with a conventional satellite, they still have complexity and exhibit all the characteristics of a normal satellite. The difference between conventional and micro or nanosatellites, besides the size is their cost (the decline in cost is achieved through a discrete and controlled increase in the project risk not only, but also because of the latent reduction of redundant elements in the satellite). These characteristics make them particularly suitable for the education and training of young scientists and engineers since, all the stages and aspects of a satellite mission must be taken into consideration [32, 33].

CubeSat

The CubeSat project started in 1999 with Jordi Puig-Suari at the Cal Poly - California Polytechnic State University with the collaboration of Robert Twiggs from Stanford University's Space Systems Development Laboratory [22]. The goal was to provide a standard picosatellite design to reduce its cost and development time. In this way, it would be possible to increase accessibility to space as well as the number and frequency of spacecraft launches. CP1 was the first CubeSat to be launched and it is represented in Figure 1.3 [34, 35].

Currently, more than 800 CubeSats have been launched. In fact, nanosatellites have become increasingly popular over the past two decades and CubeSats, in a distinctively way, are no exception. The predictions of nanosatellites to be launched in 2018 are of up to 400 units, reaching 703 units ahead in 2023, an increase of approximately 175% in a short period [24, 36]. More data on this can be found in Appendix A.

Figure 1.3: CP1 - First CubeSat built by Cal Poly students (image from Ref. [37]).

1.2.4 Spacecraft Configurations

According to the overall mission, the spacecraft will use a particular payload[2] and bus[3]. Bus tasks include: final orbit adjustments; orbit maintenance; commanding; data storage and downlink; power generation; structural support and thermal control [21].

The payload is the main driver of the spacecraft's overall design. Nevertheless, other parts need to be taken into consideration, including the mass of the all spacecraft; the power consumption; cost; the global schedule; the spacecraft lifetime and reliability and finally, the orbit. All the subsystems shall be designed to meet the requirements established [21].

For spacecraft configuration and architecture, a wide variety of features need to be evaluated and specified. Example of these which include the attitude and control technique, the use of propulsion, the solar array configuration, the communication antenna or the spacecraft autonomy. Figure 1.4 shows a CubeSat without propulsion and with three-axis stabilized (the most common configurations for CubeSats) [21, 39].

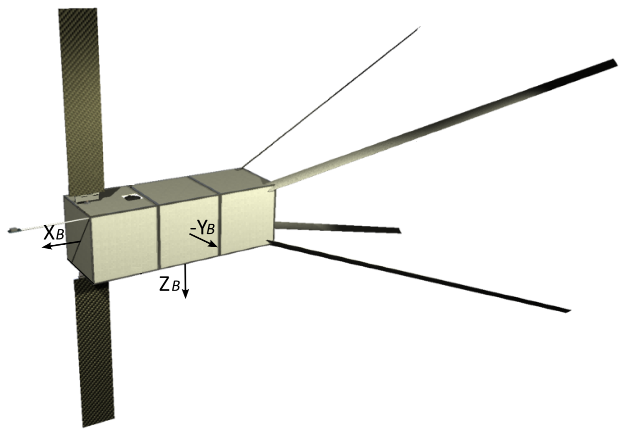

Figure 1.4: Zacube - 2, Three-Axis stabilized CubeSat (image from Ref. [39]).

CubeSat Configuration

As mentioned before, the CubeSat has a standardized configuration. The smaller version is a 10 cm cube with a mass of up to 1.33 kg and is called a CubeSat unit, "1U". Larger sizes can be built by incrementing multiples of the basic unit. At the moment 3U, 6U, 12U and 16U are usual, as shown in Figure 1.5 (Appendix A has complementary information about the type of nanosatellites launched). All of these formats must fulfil several requirements, defined at CubeSat project, which include Mechanical, Electrical, Operational and Testing requirements specified in Ref. [22].

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[2] The payload is the equipment that performs the main mission [38].

[3] The purpose of the bus, or platform is to provide all the necessary subsystems to support the payload [21].

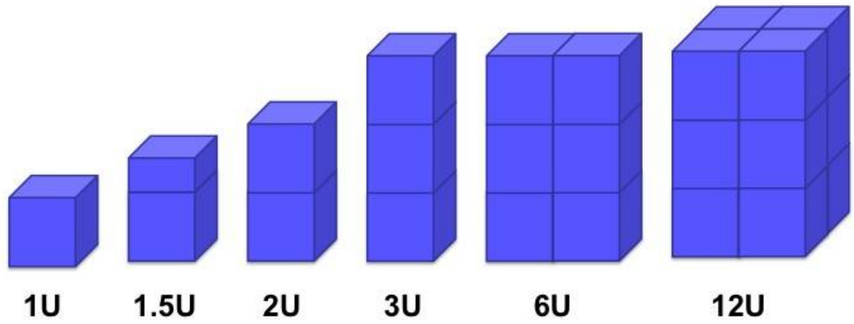

Figure 1.5: Standard CubeSat configuration (figure from Ref. [40]).

Deployer Configuration

Another important component of the mission is the deployer system. Cal Poly developed a standardized CubeSat deployer with a capacity of up to a 3U, named P-POD - Poly Picosatellite Orbital Deployer. This is widely used by CubeSats teams since it allows an easier integration with the launch vehicle. It is made of anodized aluminium and it has a mass of roughly 3 kg. Another CubeSat deployers are available however, P-POD is the most used [21, 41].

1.2.5 Spacecraft Subsystems

After the top-level configurations of the spacecraft are defined, the spacecraft can be divided into different subsystems. The following descriptions summarize the most commonly used parts that need to be developed for a final space product [42, 43].

Attitude Determination and Control Subsystem or Attitude Control Subsystem (ADC or ACS) this subsystem is responsible for detecting and controlling the vehicle's attitude;

Propulsion Subsystem (PropS) includes the spacecraft thrust, the fuel storage and plumbing, allowing the spacecraft's orbit to be controlled, as well as the de-orbit and re-entry operations (if necessary) to be achieved;

Position and Orbit Determination and Control (P&ODC) has all the sensors and software necessary to control the orbit of a spacecraft;

On Board Processing or Command and Data Handling (C&DH) consists of all the electronics and software used to receive and distribute commands, and to store payload data and spacecraft telemetry;

Telemetry, Tracking and Command (TT&C) or RF - Radio Frequency communications is the radio and associated hardware such as the antennas and cabling used for communication between the spacecraft and the ground segment;

Power Subsystem is the equipment used to generate, store and distribute the electrical energy needed for the spacecraft;

Thermal-Control Subsystem (TCS) includes all the necessary components used to control the vehicle's temperatures;

Structures and Mechanics Subsystem (S&MS) is the physical structure of the spacecraft and where every component are assembled.

A correct understanding of the space system hierarchy and its subsystems can be achieved by studying Figure 1.6.

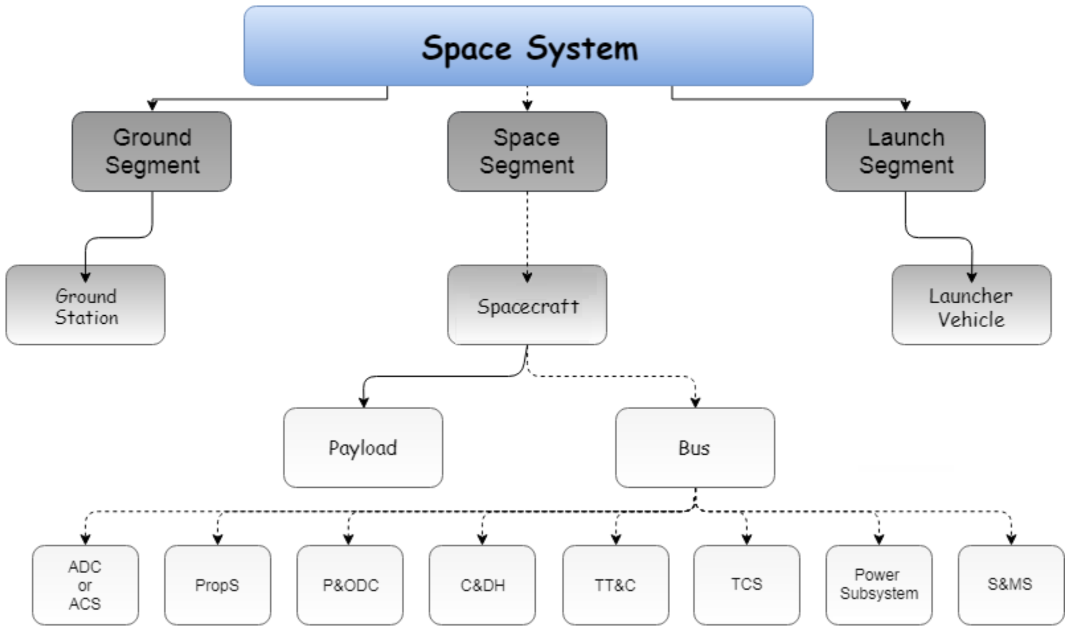

Figure 1.6: Relations between the Space system and the subsystems in a spacecraft.

1.3 MECSE Case study

"Magnetohydrodynamics/Electrohydrodynamics CubeSat Experiment (MECSE), is a student-driven project aiming to study the plasma dynamics surrounding the spacecraft when travelling in Low Ionosphere and create a benchmark for the validation of the theory that an electromagnetic field can manipulate the plasma layer. To be successful, MECSE shall orbit the Earth (LEO) gathering data on the plasma layer while using an electromagnetic generator" [44].

MECSE is a nanosatellite under development, with an expected mass of 2.8 kg, within a partnership between C-MAST - Center for Mechanical and Aerospace Science and Technologies of UBI and the nonprofit organization CEiiA.

Mission

The MECSE project has an educational objective and a few scientific and technological objectives, all of them can be consulted in Table 1.3. Aside from the mission objectives, MECSE main goal is to perform a proof of concept. Therefore, MECSE intends to demonstrate that the manipulation of plasma using electromagnetic control allows the mitigation the RF - Radio Frequency blackout that occurs when space vehicles re-enter Earth's atmosphere.

C-MAST, which is the scientific stakeholder of MECSE is also in charge of the development of the MHD - Magnetohydrodynamics (the numerical model for the manipulation of plasma), and the MHD/EHD (the device that will generate the electromagnetic field). MECSE will be the experimental test that will validate the manipulation of plasma in the travel of a spacecraft to the Mesosphere [44–46].

Table 1.3: Mission Objectives of MECSE (data from Ref. [44]).

Primary Mission Objectives		
Education	**MO1**	Provide hands-on experience to university students on space projects
Science	**MO2**	Study the formation of plasma surrounding the spacecraft when travelling in LEO
	MO3	Assess the effects of the spacecraft attitude motion on the plasma layer
	MO4	Study the effects of an electromagnetic field on the plasma layer
Secondary Mission Objectives		
Technology	**SMO1**	Develop a MHD/EHD device for plasma layer manipulation
	SMO2	Develop a modular structure for a CubeSat to be used in future space missions

Orbit

This CubeSat is planned to have a LEO - Low Earth Orbit with 350 km of altitude, an inclination of 52.6º and an orbital period of 1.52 h and a lifetime between 0.8 and 1.08 years (these and other orbit characteristics are discussed in Appendix B - MECSE Orbit) [44].

Structure

MECSE has a 3U configuration, where the payload is fitted on the first unit, while the other subsystems are mounted on the remaining units (as depicted in Figure 1.7). The MECSE configuration is presented in more detail in Appendix B - MECSE Structure.

At the current stage of the project, the primary structure was designed to be in Aluminium 7075-T6, an aluminium with zinc as the primary alloy. That material was selected because this specific type of aluminium is subjected to a thermal treatment which offers an increase in hardness along with other desired properties. This decision was reached after preliminary analyses, in particular, a linear static analysis and a thermal calculation of the spacecraft's temperature [47].

To complete the definition of MECSE's physical characteristics, it is essential to clarify some open points. The solar array deployment mechanism does not exist (for the current configuration the solar panels are attached to the CubeSat side panels). Moreover, there is no assisted propulsion nevertheless an attitude control technique is planned to be implemented. The payload is under development and therefore some of its properties and configurations, such as the antenna or its interaction with the platform, need to be analysed carefully in a near future.

Launch Vehicle

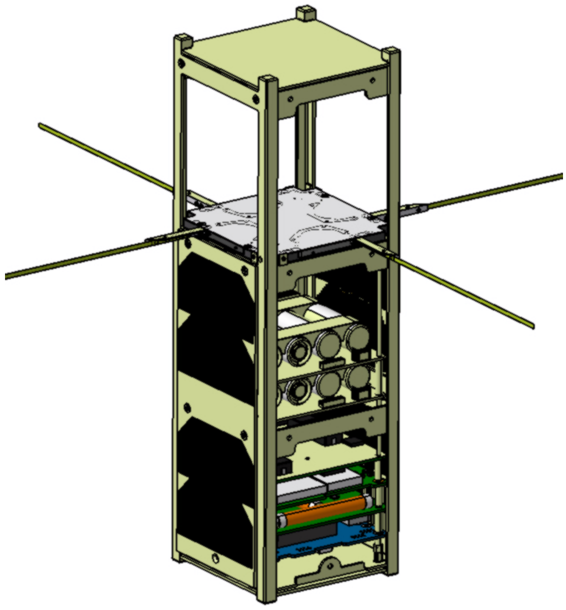

Figure 1.7: Representation of MECSE subsystems (figure from Ref. [47]).

The process of inserting a satellite into orbit is conditioned by the launch vehicle. During lift-off, the launch vehicle needs to overcome the aerodynamic forces in the lower atmosphere and the weight of the spacecraft upon itself. This stage is characterized by the maximum forces (the greatest accelerations and vehicle vibrations) in all the ascent phase [38]. Thus, this is when the satellite structure has to withstand the most critical loads and prevent them from harming or destroying critical parts of the satellite [48].

For MECSE, as a first approach the selected launcher was Vega, which belongs to Arianespace, alongside with Ariane 5 and Soyuz. Vega, like any other launch vehicle has its own specifications which need to be considered for the spacecraft design. Two of the most conspicuous ones are the launcher's profile, displayed in Figure 1.8, and the spacecraft's position inside it. They define the performance, environment and interfaces that MECSE needs to respect [49].

Requirements

In the Aerospace field, the characteristics of the satellite change depending on the requirements. Different satellites have different requirements and by consequence different configurations and different subsystems. In response to it, the verifications will also be changed. In Europe, the standards are set by ESA and can be found in ECSS - European Cooperation for Space Standardization documents [50, 51]. Contrary to the set of European norms, in the USA the requirements are defined by NASA and an overview of it can be consulted in Ref. [52].

Several requirements must be followed and verified in every subsystem of a satellite. Always matching requirements of each element, part or unit of the spacecraft and verified by computa-

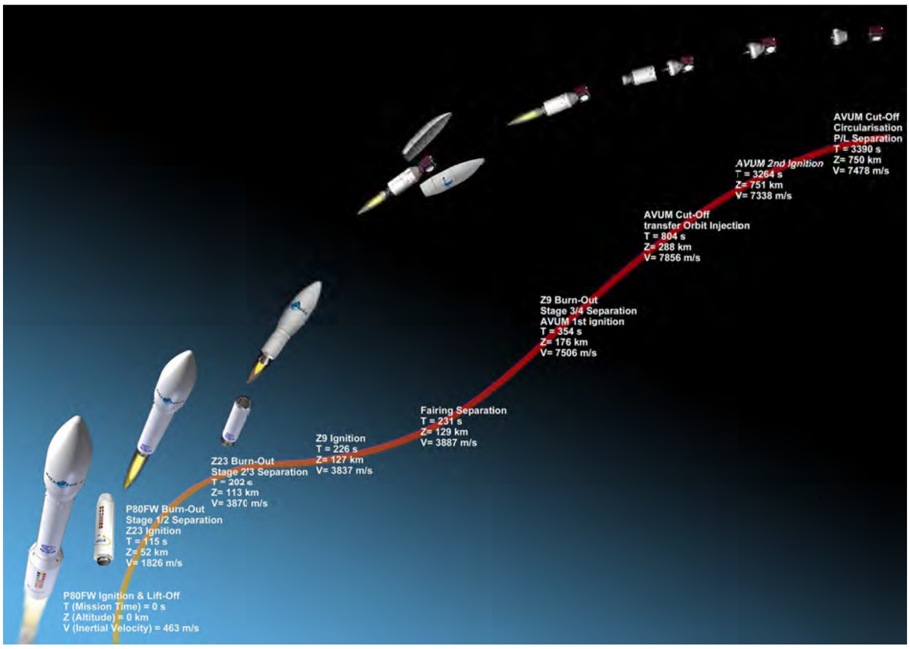

Figure 1.8: Vega typical ascent profile (figure from [49]).

tional analysis, system inspection, testing and ROD's - Reviews Of Design leading to a certified and quality product.

In inspection and testing, a significant amount of time and money is spent, finding out whether the product is capable of performing the mission for which it was designed. Therefore, the computational analysis is a much easier and cheaper way of verifying the product capabilities, so it is usually applied in the product development process [43, 53].

For the specific case of CubeSats, the basic requirements that need to be considered are explicit in Ref. [22], Ref. [52] and Ref. [53]. According to the launch vehicle, other requirements might need to be taken into account however, this topic will be discussed in chapter 2, chapter 3 and chapter 4. A simplified representation of the areas which verifications have to be performed is shown in Figure 1.9.

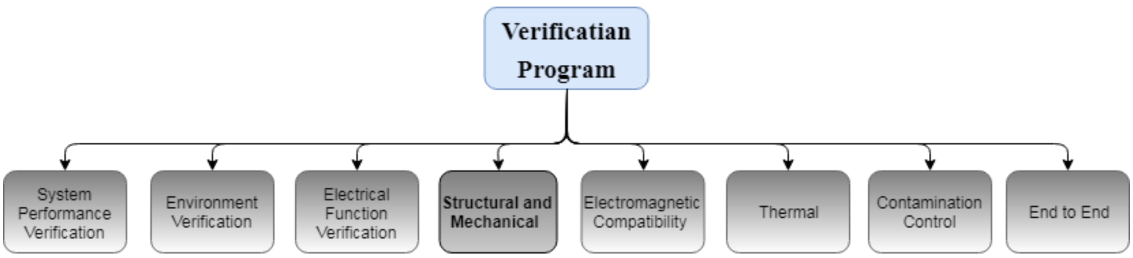

Figure 1.9: CubeSat General Verification Program (in agreement with Ref. [52]).

1.4 Objectives

The need behind this work is to certify MECSE for flight and get clearance for launch from the selected launcher.

Therefore, the added value of this work is to understand the procedures in the different stages of the product development and identify the most significant ones, in order to concentrate significant part of the resources in that process. Attempting to discover possible ways to improve standards procedures and interpret the advantages and disadvantages of the requirements from different launchers and organizations is imperative to do.

To indicate that it will always consider the most pessimist option among those studied, which will allow the application of this work in projects that have similar conditions, and will admit a possible utilization of one of the options between all the cases studied.

To accomplish the proposed goals and clarify the work strategy, the following tasks were set:

- Determine and clarify the phases for a CubeSat/nanosatellite mechanical structure development;

- Establish the necessary tasks that are essential to the development of a product and describe them;

- Identify the documentation that needs to be followed and what needs to be provided to proceed with the certification campaign;

- Analyse the performance of a group of launch vehicles and extract the more demanding conditions and present a suitable option for MECSE;

- Define the analyses and test conditions to be applied in MECSE CubeSat;

- Evaluate the most suitable phases and reviews for MECSE, in a perspective of finishing the project, in the shortest period of time possible without harming the mission performance;

- Identify the facilities conditions to perform the experimental verifications.

At the moment, the MECSE project is in Phase B, Preliminary Definition, and some documents like the PAP - Product Assurance Plan, and the AITP - Assembly, Integration and Test Plan or just Test Plan, are points to be built, or at least the baselines of these documents. Being this topic referred ahead, more precisely in subsection 2.1.2.

The intent is by the end of this work leave the structural verifications requirements outlined and the project should be in position to integrate the detailed structural design, with the realization of the defined analyses. The production of a prototype should also be prepared, in order to complete the defined tests that allow the full qualification of a CubeSat. In chapter 5, an exhaustive report of these subjects will be presented.

1.5 Dissertation Outline

This work is structured in a coherent and logical manner. The description of each chapter within this document is presented below:

Chapter 1 introduces the space exploration in a national and international point of view. It presents the purpose of this work to UBI. It also indicates fundamental themes and concepts of space scientific community. The MECSE project is explored and finally, it is presented the research objectives expected to be achieved during this study and the new contributions of this work to MECSE project.

Chapter 2 provides a first approach on several subjects, corresponding to the systems engineering discipline which is identified the project management plan, the project phasing and the norms and standards for space missions; on the structural verification process it is established the typical structural validation and dimensioning cycle using the design requirements and the mechanical environment. Afterwards a concise presentation of the verification methods is granted.

Chapter 3 refers to the tailoring of standards for the satellite structural development. Here, the set of systems engineering requirements, verification requirements, and structural requirements are formulated based on the standards and norms previously selected in chapter 2. In these requirements several points are highlighted. From the systems engineering and verification specifications are outlined the model philosophy, the test sequence, the equipment and element test levels and durations, the test tolerance and accuracy, all of it in the beginning of the chapter and later on the numerical verification requirements, alongside with the finite element analysis requirements and the correlation between analysis and tests requirements are presented. From the mechanical point of view are summarised the structural general requirements and the factors of safety.

Chapter 4 presents the evaluation of a set of launch vehicles and some of their requirements, in order to identify a suitable option to be applied to MECSE. However, it is also established the worst case scenario of mechanical environment. Afterwards, a verification approach proposal is presented.

Finally, chapter 5 presents the important steps drawn from the structural requirements and validation process of MECSE CubeSat. It also identifies the difficulties in the development of this dissertation and proposes future works to be performed by the project team.

Chapter 2

Systems Engineering and Structural Validation Process

In chapter 1 an overview of the fundamental ideas and conditions of small satellites and CubeSat's with a specification on MECSE project was presented. Furthermore, this chapter presents the fundamental concepts of Systems Engineering and Structural Validation process, containing relevant content in order to understand the different stages in the development of a satellite, the necessary tasks and documentation, indicating afterwards which are the development phases of a CubeSat Structure.

After an introduction to the basic concepts of Systems Engineering, a brief presentation of Project Management is done, close by a review of the Management Plan and a discussion about the life cycle of a satellite is subsequently made. In order to complete the chapter, a summary of the Verification process for a CubeSat structure validation is presented. Inside this topic, a more concrete approach to analyses and tests is granted.

It is important to note that in this work, the main source of information was provided for the most part from ESA. Even if NASA has very similar documentation, that could be used, comparing to the ESA standards, the NASA documentation is used for supplement and to fulfil some gaps identified during the investigation of specific topics.

The option, to use the European regulation instead of NASA regulation, was taken since the ECSS presents the necessary set of requirements in a more structured way, in order to promote the development of a spacecraft structure. Moreover, like it is highlighted in Ref. [54], ECSS presents the documentation that should be followed once the MECSE launch vehicle is intended to be launched integrated into an ESA program, which follows the ECSS standards [54].

Other organizations could be taken into consideration, like for example the ISO - International Organization for Standardization, or the Chinese documentation. The guidelines from these organizations have not been analysed in detail on a count that, even if the ISO/TC 20 standards are dedicated to aircraft and space vehicles, and it has vast and trustworthy information, the majority of their documentation is paid [55]. Important to note that, at the moment the ECSS and ISO are cooperating for the development of standards in the area of space systems and operations [56].

On the reverse side, the case of China, their records are being kept secret by the organizations that manage their space industry [57]. Another point, it is just recently China adopted regulation for their space industry, what lead to an understandable lack of information about important areas of study [58].

2.1 Systems Engineering

In every engineering project, the final product is always a result of a good design and a developing process across several sections. Due to the ever-increasing complexity in the development of projects and with the intention to reduce the risks of the product life cycle, and for a more efficient projection, control of expenses and schedules the systems engineering approach has been created [59].

The systems engineering is an interdisciplinary approach, with a huge range of applications. It has the main fields of study the performance, the cost and schedule, the training and support, the manufacturing, the tests, the operations and the disposal [60].

Being such a multifaceted way to feature so many subjects, has also the ability to transform the requirements and the desired function into a final product able to achieve the designed mission. It is an iterative process during the whole life of the product but, it has its predominant influence in the initial phases of the project where the TS - Technical requirements Specification defines the approach that the systems engineering function needs to complete [61].

Apart from it, the systems engineering discipline has the responsibility to define the PMP - Project Management Plan which is included in the SEP - System Engineering Plan. Further that, it has the responsibility of identifying critical items in cooperation with the PA - Product Assurance. Making possible after the definition of the critical items the allocation of requirements, specifications, tasks and schedules for all phases of the development of a product [62].

The SEP defines the approach, the method, the procedures, the resources and the organization that will coordinate and manage all technical activities that specify, design, verify, operate and maintain a project in conformance with the mission that was proposed initially, since the lower level (a single part) to the top-level (the space segment), establishing not only but also the product and function tree for each item present in the spacecraft [61, 62].

Figure 2.1 shows the disciplines that have a contribution to the evaluation of a product and the systems engineering sub-functions, like it is the systems engineering integration and control, the analysis, the engineering requirements, alongside with, the design and configuration and the verification and validation. In Appendix C - Systems Engineering, Figure C.1 the interactions between the various sub-functions of systems engineering are reported.

2.1.1 Management Plan

The PMP is a way to define the methodology and organize the procedures to generate a product, in this case, will be applied to a satellite.

When the intention is to understand the PMP and how it is set up, some other documentation need to be known first. The PRD - Project Requirements Documents that typically comprise the statements of work and the technical requirements documents (which within has the technical requirement's specification [63], management requirements [64–67], engineering requirements [62, 68], product assurance requirements [69], programmatic requirements, tender requirements and other

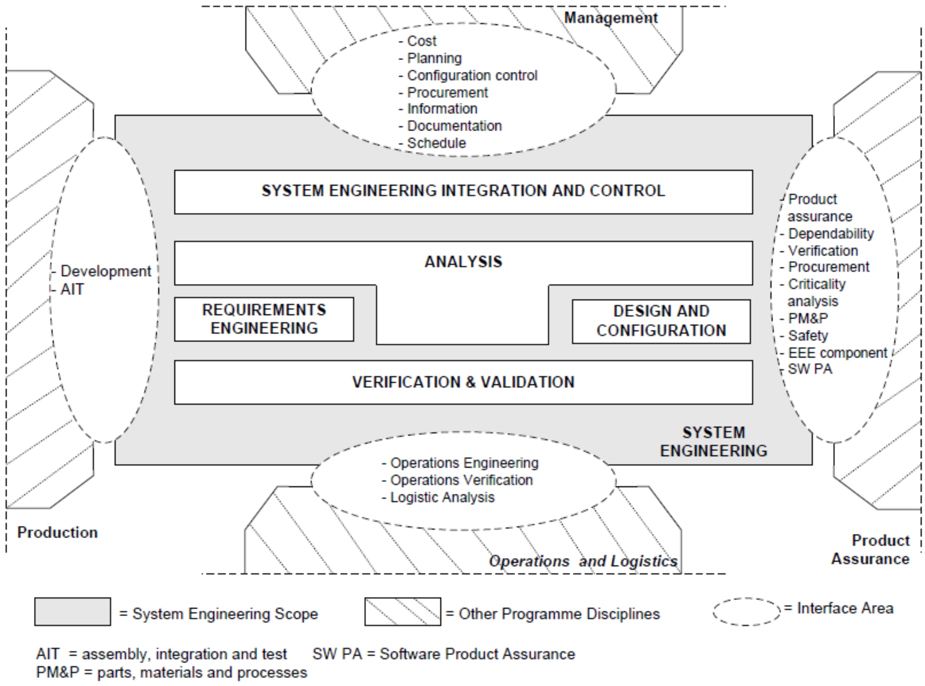

Figure 2.1: Systems Engineering fields of study (figure from Ref. [62]).

project specific requirements) are a crucial part of the PMP [70].

The PBS - Project Breakdown Structure another critical document in the PMP, provides the basic lines for creating a common understanding between all actors in the chain of a product. It is obtained with the study of the function tree, the specification tree, the product tree, the WBS - Work Breakdown Structure, among others documents [70]. The PMP alongside with the PBS and the WBS of MECSE can be consulted in Ref. [54].

For more information and some definitions about the topics referred previously, consult Refs. [60–62, 70], it indicates how to build and how to organize the different specifications refereed previously.

2.1.2 Project Phasing

To help in understanding how a Space product is constructed this topic has been added. With the aim of discussing some phases of a product developing, so that later on this chapter 2 and in chapter 3 a more in-depth argue about the pertinent and relevant phases of the structural development of a satellite is made.

Every project is divided into phases, with the aim of defining which are the important task in the development of a product, but also, identifying the decision points to overcome and the different reviews that allow the project to advance.

Below, a clarification about the different phases of a space project will be done following the ECSS standards, Ref. [70]. For further information and in agreement with NASA guidelines, the NASA instructions for project phasing consult Ref. [59].

Phase A - Feasibility

In phase A, Feasibility, a preliminary SEP, PMP, and PAP is performed; an elaboration of possible systems, operation concepts, architectures and an equilibrium between them is necessary to be achieved. A definition of the function tree and the risk assessment is another goal, alongside the evaluation of the technical and programmatic feasibility of possible concepts. The identification, quantification and characterization of critical technologies and elements for technical and economic feasibility are assignments to do. To finalize, it proposes operations concepts and technical solutions for the system.

In this early stage of the product development, the PRR - Preliminary Requirements Review is the review to be reached.

Phase B - Preliminary Definition

This phase is one of the most extended ones, where a large range of assignments need to be accomplished. In the preliminary definition, the finalization of the PMP, SEP and PAP is a task to do. Moreover, the establishment of the baselines for cost and schedule; the preliminary OBS - Organizational Breakdown Structure; the confirmation of the technical solutions for the system concept; the operations concepts and their feasibility with respect to programmatic constraints is done. The conduction of trade-off studies for the selection of the preferred system concept and a technical solution is also a concern of the preliminary definition. When the preliminary design is achieved the construction of the verification program including the model philosophy is delineated.

The identification and definition of external interfaces; the initiation of the pre-development work on critical technologies, as well as long-lead items ordered to reduce the development risks and meet the project schedules are functions to be performed in this phase. The preparation of the mitigation plan and the disposal plan; the conduction of reliability and safety assessment and to finalize the product tree, the work breakdown structure and the specification tree are the others responsibilities of Phase B.

The reviews in this phase are the SRR - System Requirements Review, and PDR - Preliminary Design Review are the reports to be fulfilled in this stage.

Phase C - Detailed Definition

The Detailed definition phase is responsible for the detailed design definition of the system at all levels. It is also responsible for the production, testing development and pre-qualification of critical elements and components formerly selected; dictating the production and experimental tests on physic models; the AITP for the system and its constituent; the detailed definition of the internal and external interfaces. Phase C has the mission to question the preliminary user manual and the risk assessment. A simulation of the system performance is often created in order to increase the confidence in the established design or identify issues on the product in development.

The CDR - Critical Design Review is the review operated at the end of the detailed definition and it is the most important document in the design stage.

Phase D - Qualification and Production

The tasks for this phase can be described like the fulfilment of the qualification testing and all the necessaries verifications activities; finish the manufacturing, assembly and testing of flight hardware/software and it culminates with operation tests between space and ground segment and with the preparation of the acceptance phase.

In this phase, Qualification and Production the reviews are extensive and expensive with a major incidence in the QR - Qualification Review. The AR - Acceptance Review and the ORR - Operational Readiness Review are the others reviews to be performed. Complementary information about the documents and reviews expressed previously consult section C.1.

Figure 2.2 represents in a visual way the project development life cycle and the relationship between the phases and the reviews.

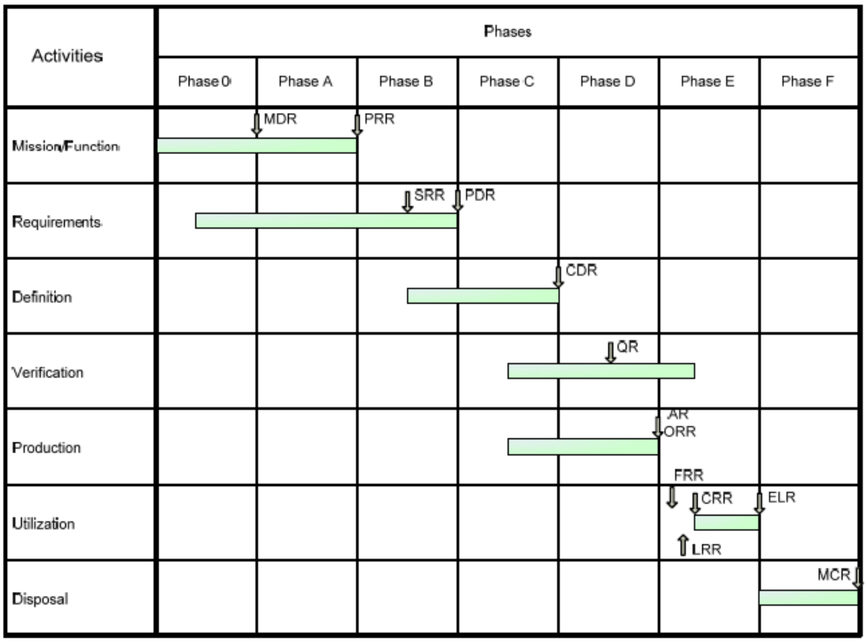

Figure 2.2: ESA Typical project life cycle (figure form [70]).

Along whit the documents described in the description of the various phases, other essentials documents are indispensable to be compiled, the ICD's - Interface Control Documents[1], or the SOW - Statement Of Work[2], are just some examples.

2.2 Norms and Standards for space missions

The creation of ECSS, in 1993 aimed to establish a coherent, single and user-friendly set of norms and standards to apply in all European space activities [71].

[1]ICD's define the external and internal interfaces between all the subsystems of a space vehicle [60].

[2]SOW defines the allocation of specific engineering requirements per phase depending on the business agreement specific document [62].

This uniformity was compiled in the search for a more efficient safety in space products and projects, simultaneous with the possibility of increasing the competition in the space industry. It uses a range of standards, in vast areas of the project for a definition of the tasks to be performed [71].

Following the ECSS documents does not just help to identify which are the subjects that should be approached but also, support the life cycle development of a product. Including management subjects and practical areas from the standardization of facilitates, to the communication between all parts involved in the project (reducing risks and ensuring the interface compatibility between all the teams, and consequently with all the subsystems, allowing a better cost-effectiveness in space programmes) are just two examples of it [71].

The aspects that are contemplated, in ECSS are organized into four major categories: the Project Management indicated in ECSS standards by the abbreviation, ECSS - M; the Engineering category pointed out like, ECSS - E; the Product Assurance area outlined by ECSS - Q and the most recently created, the Sustainability category, ECSS - U [72].

More than the definition into categories, also three types of documents were formulated. The Standards (ECSS - CT[3] - ST, the most common documentation in ECSS and the one that prevails in case of conflict with some other ECSS document [73]) with information about the verifiable requirements of each category; the Handbooks (ECSS - CT - HB) to provide background information, orientation, advices and good practices related to the discipline in question; and the Technical Memoranda (ECSS - CT - TM) to present data which are not subject for a standard or handbook or not yet mature enough to be published in one of the others ECSSS documents [72].

The different disciplines, and the major standards of each discipline can be reviewed in Figure 2.3.

2.2.1 Space engineering branch

This work focuses on the engineering and mechanical program, but even in the engineering program the complexity is high and the disciplines to be approached are abundant. In Appendix D - Space Engineering Branch, a better understanding can be reached, with an identification of the different types of documents in the space engineering department and also in the mechanical discipline.

Therefore, a preliminary selection of the most important documents for this dissertation has been done. From Figure 2.3 was selected the E-10 discipline, Systems Engineering and the E-30 discipline, Mechanical Engineering [71]. Moreover, the selected standards can be consulted in Figure 2.4 and the handbooks and thecnical memorandas in Figure 2.5.

E-10 Systems Engineering

The E-10 discipline has the function, such as the systems engineering, of defining a large range of management requirements in several areas of space systems or product development. This discipline has several topics of interest for this work, that can be seen in the following description.

[3]CT - ECSS Category, represents the categories in the ECSS documentation. It has the possibility to be referred to M - Management, E - Engineering, Q - Product Assurance or U - Sustainability.

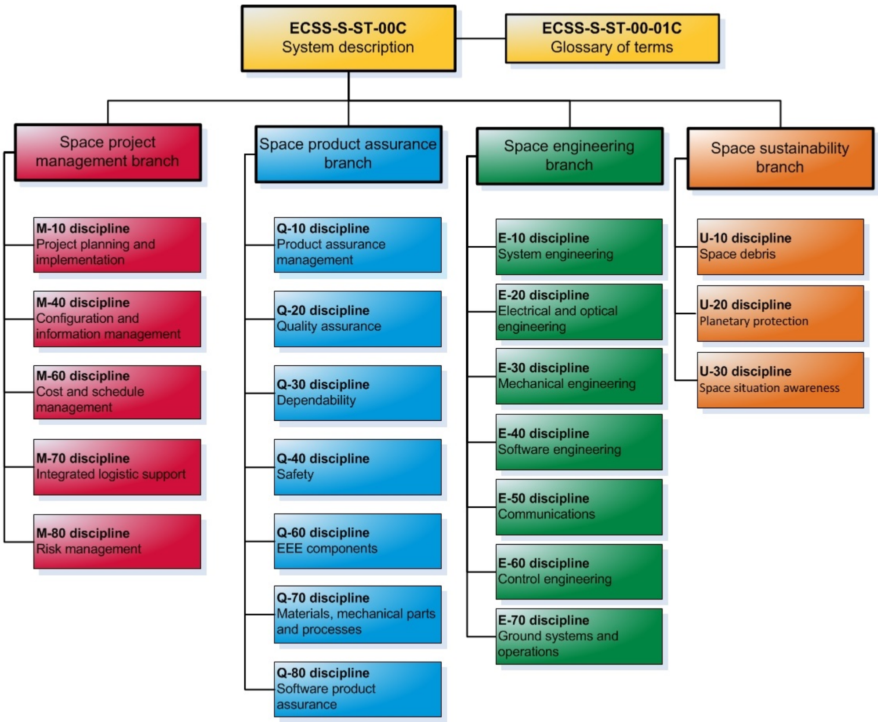

Figure 2.3: ECSS Disciplines, System Description (figure from Ref. [73]).

The Systems Engineering general requirements (ECSS-E-ST-10C Rev.1) was selected once it provides information about the tasks, objectives and ways to implement systems engineering requirements. This organization and management of the tasks to perform allow the minimization of technical risks and by consequence the costs for space products. This topic has already been concisely reported in the previous section 2.1 [62].

The standards Verifications (ECSS-E-ST-10-02C Rev.1) and Testing (ECSS-E-ST-10-03C Rev.1) are alongside with the handbooks Verifications guidelines (ECSS-E-HB-10-02A), Testing handbook (ECSS-E-HB-10-03A) and the handbook, TRL - Technology Readiness Level guideline (ECSS-E-HB-11A) pertinent documentation to be studied. Since, it allows a deeper knowledge of the verification process, strategy and the specific requirements for the verification programme implementation into a spacecraft [50, 74–76].

The Space Environment (ECSS-E-ST-10-04C) is applied to all products that are planned to operate in space. It defines the external physical environment, the induced space environment, and other elements identified by former space activities [77].

The Technical requirements specifications (ECSS-E-ST-10-06C) has an important purpose in this work once, it defines the different types of requirements but also provides an overview of the technical requirement specification and additionally presents the way that technical requirements should be formulated [63].

The Reference coordinate system standard (ECSS-E-ST-10-09C) indicates the mutual interrelationships and transformations, used to define reference coordinate systems from different components, parts or even between the space segment and others segments in the space system [78].

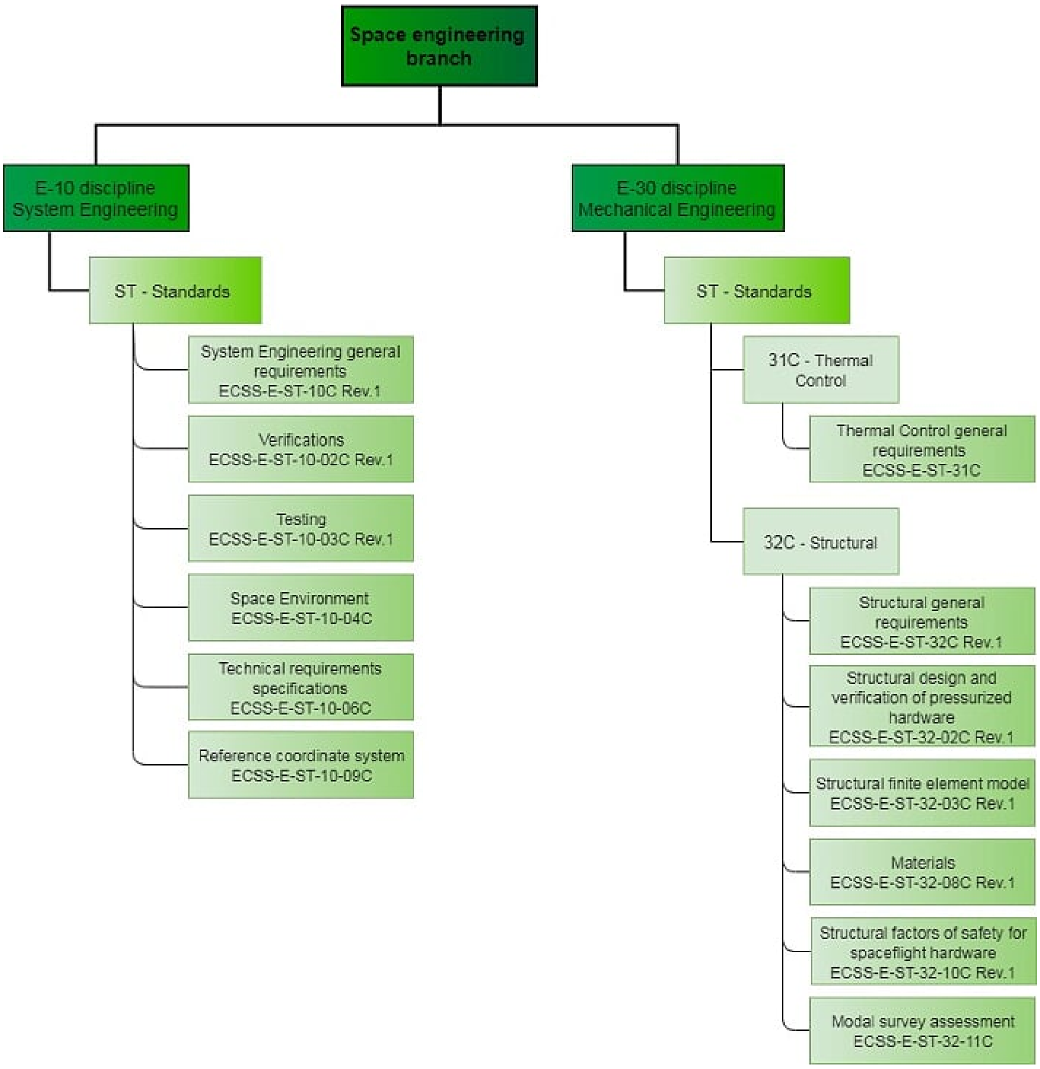

Figure 2.4: ECSS Space Engineering branch, selected Standards (figure adapted from Ref. [71]).

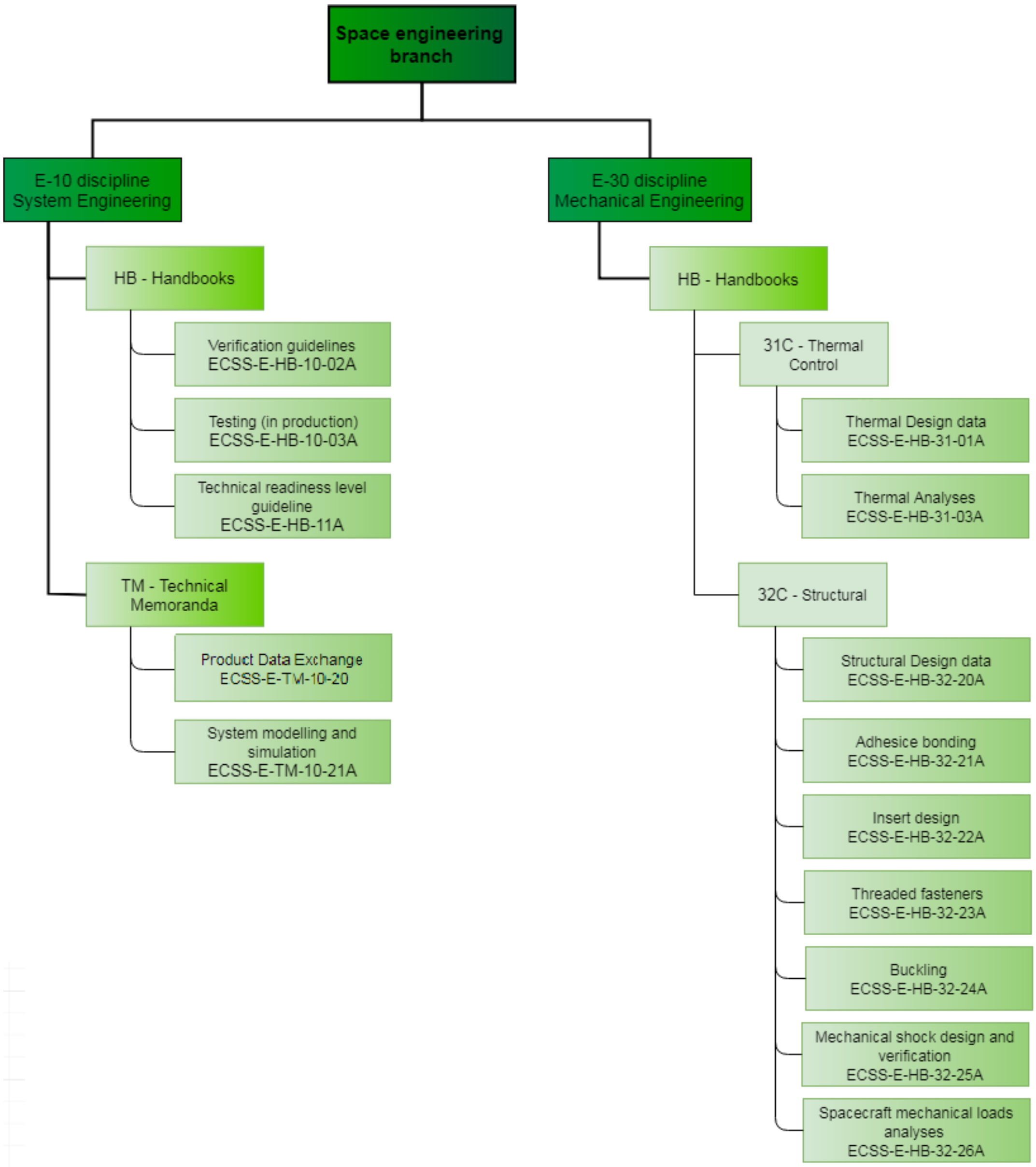

Figure 2.5: ECSS Space Engineering branch, selected Handbooks and Technical Memoranda (figure adapted from Ref. [71]).

E-31C Thermal control

The Thermal control discipline has the goal of giving a description of the thermal conditions that a spacecraft need to overcome in order to successfully complete the mission. It classifies the temperatures from the ground and pre-launch phase until the end of the life cycle of a space product [79].

The Thermal Control general requirements standard (ECSS-E-ST-31C) is a definition of the requirement outlined for the temperature range of the satellite [79]. The handbooks, Thermal Design handbook (ECSS-E-HB-31-01A) and the Thermal Analyses handbook (ECSS-E-HB-31C-03A) are an important complementary section of this standard. Once the goals of these handbooks are providing practical guidance to the thermal engineers for an efficient and high-quality verification of the environment condition, at the same time that support and recognize the interconnection between the system the environment and the interferences between the different units at a space product [80, 81].

E-32C Structural

The discipline E-32 defines all the Structural needs referred to the Mechanical engineering and represents a set of conditions that need to be applied to every satellite [82].

The Structural general requirements (ECSS-E-ST-32C Rev.1) defines requirements that need to be followed in all life cycle of a satellite, like the definition and specification of the design, the development, the verification, the production, an eventual disposal, among others [82].

The Structural design and verification of pressurized hardware standard (ECSS-E-ST-32-02C Rev.1) covers themes such as the structural design verifications of metallic and non-metallic pressurized hardware (which includes valves, pumps, lines, fittings and hoses), as well as special pressurized equipment (batteries, heat pipes, cryostat, sealed containers are some examples). It is analysed with the goal of demonstrating the qualification of the design and the performance, of these parts but also subjecting the flight hardware to experimental verification in order to confirm that it is free from manufacturing defects [83].

The Structural FEM - Finite Element Model standard (ECSS-E-ST-32-03C) has the function of defining the requirements for FEM. In the structural FEM requirements are included the necessary checks to be performed, and the parameters to be fulfilled in order to assure the numerical model quality [84]. The handbooks Buckling of structures (ECSS-E-HB-32-24A), Mechanical shock design and verification (ECSS-E-HB-32-25A), and Spacecraft mechanical loads analysis (ECSS-E-HB-32-26A), assist the standard referred previously. If the handbooks do not define requirements about analysis or models, its present guidelines, suggestions and good practice advices for design, analyses and verifications. It is a meaningful help for engineers involved in tasks of verification and qualification of equipment and sub-systems inside a satellite structure or yet in the element [85–87].

The Materials standard (ECSS-E-ST-32-08C) specifies the mechanical engineering requirements for materials, but besides that, this standard also defines requirements and verifications (including destructive and non-destructive tests) for mechanical and physical properties of a material to be used in space applications [88]. The handbook Structural Design data (ECSS-E-HB-32-20A, constituted by eight parts, and also known as, Structural materials Handbook) and the handbook Insert design (ECSS-E-HB-32-22A) are an extensive data input for design engineers, combining the materials characteristics with design aspects, along with factors associated with verification, joining and manufacturing [89, 90].

The topic of design and verification of joints by metallic or composite materials are discussed in more detail in the handbooks Adhesive bonding (ECSS-E-HB-32-21A), and Threaded fasteners (ECSS-E-HB-32-23A) [91, 92].

The Structural factors of safety for spaceflight hardware standard (ECSS-E-ST-32-10C Rev.1) defines the FOS - Factor Of Safety and other margins to be used in dimensioning, design and verifications of spaceflight hardware for qualification and acceptance [93]. This standard is always used in conjunction with other standards like the Thermal Control general requirements, the Structural general requirements and the Structural design and verification of pressurized hardware.

The Modal survey assessment standard (ECCS-E-ST-32-11C) specifies the requirements on a modal survey test; involving the preparation, the execution and the evaluation of the test in comparison with the dynamic analyses to be performed previously. This standard is an important source of information for the experimental verification responsible engineer [94]. In agreement with the previous standards, and using other references as support, an integration of the fundamental concepts of a structural verification process is presented next.

2.3 Structural verification process for small satellite

"Verification proves the product is right. Validation proves it is the right product" [75].

The verification process of any space vehicle has three phases, with the aim to demonstrate that the design can withstand the specified requirements, and present to the launcher authority the functionalities and capacities of the spacecraft. It grants the integrity and the performance of the project. With the confirmation that the overall system is able to fulfil the mission requirements [53].

One of the phases is the Qualification phase, and it intends to qualify the design and the performance of the product. The other parts of the verification process are the Acceptance phase, and the Pre-launch phase. The Acceptance phase intents to guaranteed that the product is free from manufacturing defects, in agreement with the qualification design. And, the Pre-launch phase confirms that the product is capable of functioning as planned during launch, and in early stage operations [50].

The verification process can be applied to all levels of assembly, from the unit or product up to the entire system - satellite depending on the type of the verification that is intended to be performed. Commonly, tests (experimental verifications), analyses (numerical or analytical verifications), ROD's, and inspections, in specific phases of the project are the methods to demonstrate the capacities of a satellite [50].

In Figure 2.6 is introduced a typical structural design validation cycle, and it is clear all the stages until the manufacturing and operation of a spacecraft structure. Alongside with the structural verifications, that lead to the structural validation. Other subsystems of the spacecraft need to overcome their verification process, as mentioned in section 1.3 - Requirements. Just to present an example of different verification programs, aside from the structural, a generic overview of the Software verification can be consulted in Ref. [95] and the Electromagnetic compatibility, that will be necessary to apply in MECSE payload is particularised in Ref. [96].

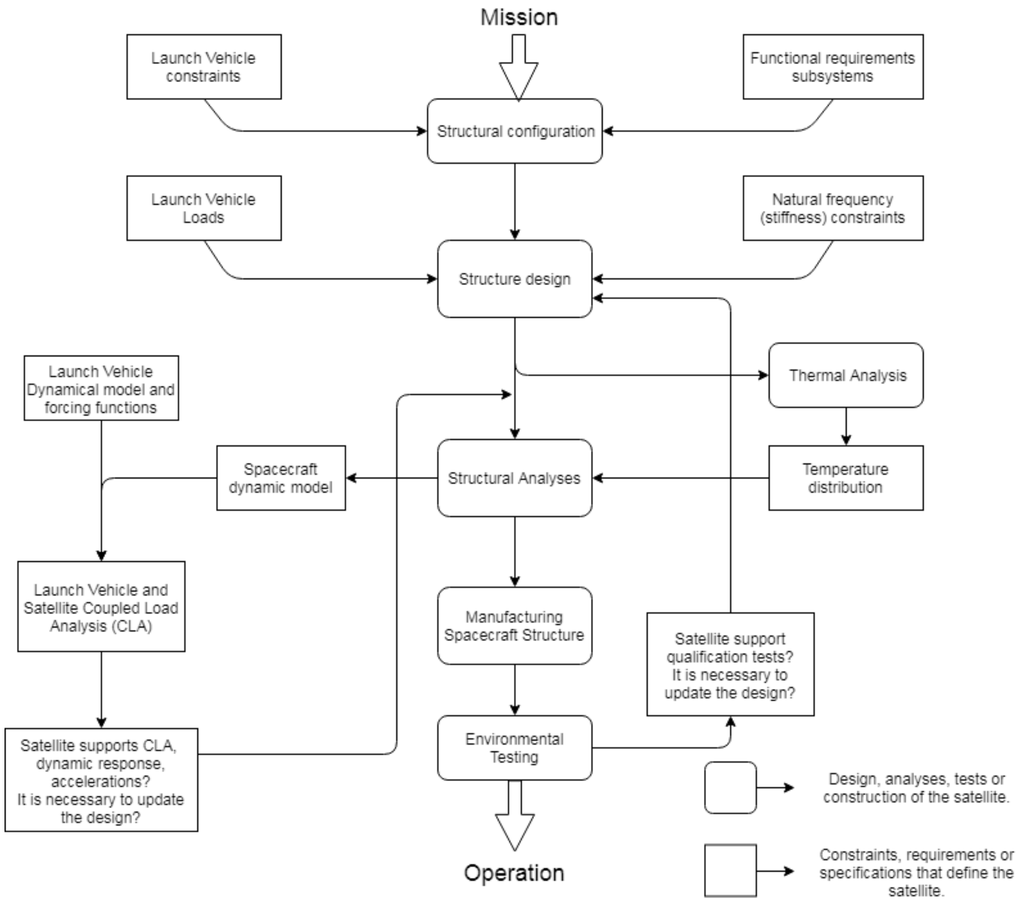

Figure 2.6: Typical Structural design validation cycle (figure in agreement with Ref. [17]).

The typical design validation cycle of every sub-system is always in agreement with the VP - Verification Plan (a flow diagram of this topic is shown in Appendix D - Verification Plan), which most of the times is combined with the AITP - Assembly, Integration and Test Plan or just Test Plan producing a single document, the AIVP - Assembly, Integration and Verification Plan for a better understanding of what is asked to all the partners, in every situation inside the verification process [50].

Other substantial documents are the VCD - Verification Control Document, the tests specifications and test procedures, the reports from analysis, tests, inspections and ROD's. To indicate that all the verification reports need to pass over the VCB - Verification Control Board, for a final review. A complete idea and approach about this matters can be consulted in Ref. [50].

From Figure 2.6 some points can be highlighted. The launch vehicle requirements will be particularly relevant, since it defines many of the satellite constraints; the spacecraft configuration and the preliminary design are the baselines to the development of the product (discussed in Design Requirements); the manufacturing of the structure is the process to achieve after the structural analyses has been performed successfully; and the structure validation, will be granted when the environmental tests are performed. Specific aspects of these last two themes will be debated in chapter 3 and chapter 4.

Design requirements

The preliminary design is defined by the top-level requirements, which characterize the leading functions to be executed and it should be the amplest possible, allowing the designers to think in multiple approaches conducing to large trade-off and the most sustainable and optimized solution as possible.

The Mechanical design specifications, that allow the development of a preliminary structure to a detailed and final structure for a CubeSat, are mainly imposed by the launch vehicle, the mission and the orbit (the descent and landing are also requirements to have in mind if the spacecraft is designed to re-enter the earth) [60].

Some of the main specifications and requirements are cross-referenced to all satellites, such as the mechanical environment; the minimum admitted resonance; the static loads; the sine specifications; the factors of safety. These requirements will directly affect the stiffness and by consequence the mass and the gravity centre of the satellite.

Figure 2.7 defines the requirements for the design of structural components of a spacecraft. It contemplates two of the points expressed in Figure 2.6, the structure design and the structural analyses, and it covers the lower level topics on this theme - Design requirements.

Mechanical Environment

The mechanical structure of a satellite must be designed to withstand all the loads and support all the key components during the entire life of a space product. The life cycle of a space product can be distributed in different phases, one from manufacturing to the transportation - Ground environment other on launch - Launch environment, and in the end the operational mission - Space environment (in some cases also the re-entry is a moment to be considered). During all these steps, the environment surrounding the satellite has properties completely distinct and all of them have to be taken into consideration.

The launch is defined by high loads in all the directions, just like the highest vibrations (it was already approached in section 1.3 - Launch Vehicle). When the satellite is at space, the thermal requirements of the structure are extreme because of the big thermal variations in short periods of time. The ground environment can not be neglected once the forces applied in transport and handling can reach significant values that could harm the satellite [42, 43, 97].

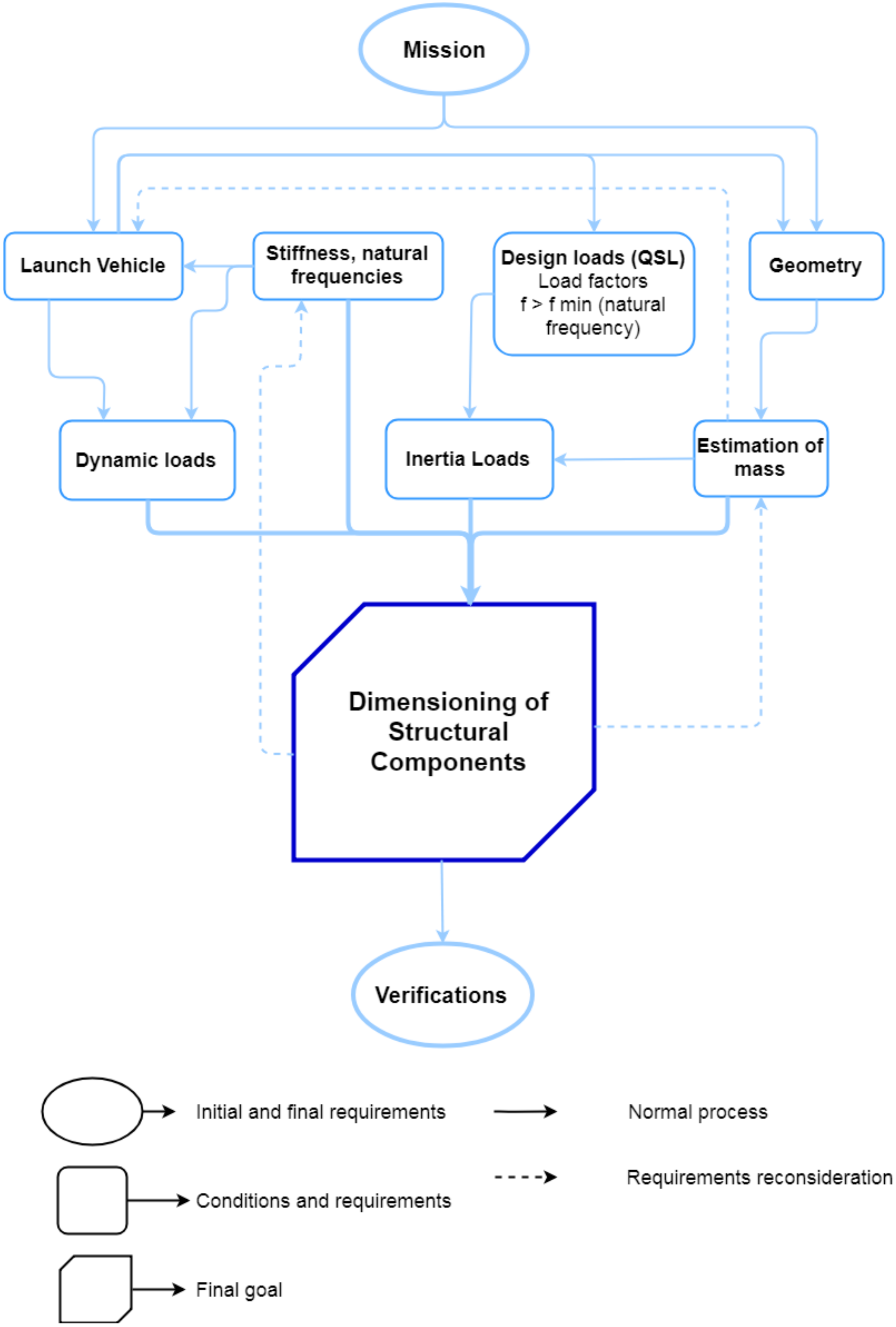

Figure 2.7: Sizing of structural components (figure in agreement with Ref. [43]).

In every single project, the WCA - Worst Case Analysis is always taken into consideration, and this work will not be an exception, once it assumes that all components of the system are at their most pessimist operational state. If the subsystem can overcome the WCA conditions, then it can be assumed that the structure will succeed in the other load cases.

Due to the different launch vehicle stages that transport the payload into the final orbit, various types of loads need to be taken into consideration [43]. More data about the occurrence of different types of mechanical loads, in launch vehicles, are revealed in Appendix E and can be consulted in Ref. [98, 99].

There is a big variety of loads categorization, just for example, exist a category according to the environment, a category according to the frequency of the load, and it is the one listed below, among other [17, 43]. A visual representation of the different types of loads and their frequencies can be consulted in Figure 2.8.

- Static loads;
- Steady State Accelerations;
  - QSL - Quasi-Static Loads;
- Low Frequency Accelerations;
  - Sine Vibration;
- Broadband Vibrations;
  - Acoustic loads;
  - Random Vibrations;
- Hight Frequency Vibrations;
  - Vibration Shock.

The values that represent the WCS are the design limit load. But to obtain MoS - Margins of Safety an ultimate load is implemented for a spacecraft qualification and acceptance. The ultimate load is achieved by increasing the design load limits by a constant.

The FOS used in the aerospace industry are really vast and each organization has their own conditions, just like the analyses or tests required. An overview of the FOS and MoS is consulted in Ref. [68, 93].

It is important to refer that according to the information provided from Ref. [22] the CubeSat for acceptance and launch approval, should perform analyses and tests with information from the launch's vehicle mission manager, since it is the one that gives more accurate values about their missions and will decide if the approval for launch is granted or not.

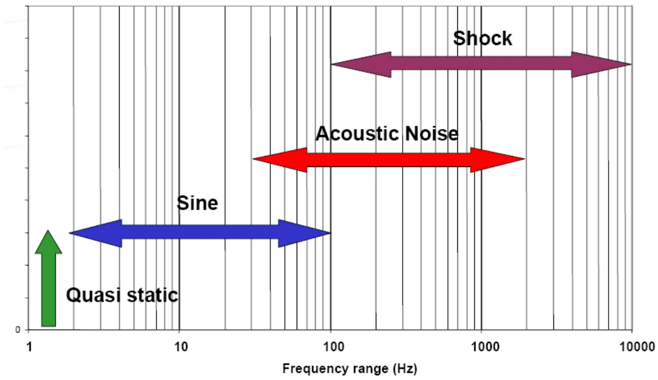

Figure 2.8: Dynamic environment specification (figure from Ref. [48]).

2.3.1 Analyses - Numerical Verification

One method of analysis can be the numerical approach, and it is executed as a complementary or alternative form of experimental testing. Although tests are the most reliable form of verification due to their high cost and time-consuming, the analyses have been captivating a substantial part of the verification programme [42].

The analytical verification is appropriate for early stages of the product development, once it allows quick changes in the components that are being considered, and also grant the possibility of the whole spacecraft structure be analysed before any hardware is manufactured [50].

With these characteristics, the qualification of parts, unit or even subsystems without doing a simple test is possible [17]. Aside from it is important to note that qualification of a product by analysis can also be done using a process of similarity, but a complete acceptance of a spacecraft only is viable using a model with experimental tests [100, 101].

Apart from the similarity, the verification by analysis shall consist in performing theoretical or empirical evaluations using techniques that comprise modelling and computational simulations, with statistical and qualitative design analysis [75]. This theme will be discussed in more detail in chapter 3.

The analyses is divided into two major categories, one is the static analyses (strength) and the other is the dynamic analyses (vibration, stiffness). These analyses are done with the purpose of obtaining a verification of all the stresses, strains, bending, shearing, and other structural demands.

According to Ref. [50, 99] the typical structural analyses that should be performed as part of the verification process are:

- Static and Linear Stress Analysis;
- Modal Analysis, to verify the frequency requirements;
- Harmonic Response Analysis, to simulate time-domain loads, to validate mathematical models and to simulate sine tests;
- Random Vibration Analysis, to predict the spacecraft response to random environment;

- Acoustic noise Analysis, to check spacecraft response to broadband vibrations and to derive the random spectra;
- Fatigue and Crack growth analysis, to verify the life cycle of critical structural elements;
- Stability Bucking Analysis;
- Thermal-elastic analysis, to evaluate the thermal distribution along the structure;

A brief explanation of the most important analyses to be performed and why they are a priority for a satellite structure is detailed below.

Static and Linear Stress Analysis calculates the effects of steady loads, while ignores inertia and damping effect. The focus of static analyses are the displacement, stress, strains and internal forces on a structure, caused by static loads. In this analysis, it is considered that the structure has a slow response with respect to time and it is important to be applied on the ground, transportation and handling loads, with the aim in determining the MoS [102, 103].

Modal Analysis is one of the mandatory tests to be performed. It is used to evaluate the vibration characteristics (natural frequencies and mode shapes) of a structure, while it is being designed. If the natural frequencies of the satellite matche with the natural frequencies of the deployer system or launch vehicle, then resonance takes place and it can not happen, once resonance is a critical response in any type structure. The modal analysis is also used to validate the dynamic mathematical model and it is achieved using the correlation between test results and analysis results. If both results values match or at least the order of magnitude match, the dynamic structural analysis is valid [48, 103]. This subject is presented in detail in section 3.2.

Harmonic Response Analysis is a technique used to determine the steady-state response of a linear structure to loads that vary sinusoidally (harmonically) with time. This analysis technique calculates only steady-state forced vibrations of a structure. Transient vibrations, which occur at the beginning of the excitation, are not accounted for, in a harmonic response analysis. The peak harmonic response occurs at forcing frequencies that match the natural frequencies of the structure (resonance frequencies), being possible one more time to validate the mathematical model with experimental tests [42, 104].

Random Vibration Analysis is performed when the input load cannot be defined in a specific time or frequency but such physic phenomenon is necessary to be studied. Conditions like the rocket motor vibrations, engine thrust or vibrations resultants from acoustic loads are some sources of random vibrations loads. It is important to be studied in a spacecraft, due to the enforced acceleration at the interface of the space vehicle and the launch vehicle and between elements inside a satellite. The random vibration will carry consequences into the damping of the structure, and consequently in the passive and/or active damping corrections to be adopted [105–107].

Stability Bucking Analysis is a linear static analysis, and the structure is normally considered to be in a state of stable equilibrium. When the applied load is removed, the structure is assumed to return to its original position. However, under certain combinations of loading (if the load exceeds the value of the ultimate load for buckling, DBL - Design Buckling Load), the structure may not support the loads and continues to deflect without an increase in the magnitude of the loading. In that case, the structure has actually buckled or it has become unstable, this analysis will demonstrate what is the ultimate load for the satellite structure that should never occur in real life of a product [85, 102].

Thermo-Elastic Analysis is performed to check the thermal deformation and stress due to temperature gradients in the structure. It must be calculated to check alignment requirements. The thermal stress is not of extreme importance, but the thermal distribution in the spacecraft has to be considered since, the definition of the location of particular parts of the satellite will take this theme into account [81].

2.3.2 Test - Experimental Verification

Experimental verifications are currently performed to assure that the mathematical model is right and accurate and it is done in critical elements of the product. Aside from it, the main purpose of performing structural tests is to qualify the spacecraft structure and afterwards it can be accepted for flight, in agreement with the AITP [50].

From AITP some are the documents that support it, aside from the test procedures are often included the test block, the expected results and the test values, that indicate the success or failure of the test.

In a real life cycle of a product, many of the loads occur simultaneously, but yet any test is able to simulate all the loads at the same time and the analyses would be much more difficult to perform, if it happens. In consequence of these points, the tests are split into various loads categories in agreement with what was presented in section 2.3 - Mechanical Environment.

It is always necessary to be alert to a circumstance, the over testing. It can occur when an element or system is tested more than one time, for the same type of load condition or for a set of different types of load conditions. But it can also happen when the sine vibration test or the random vibration test is performed, and the accuracy of the induced loads are not properly controlled, what could lead to an excess in the test load intensity. If one of these cases happen the result could be representing inadequately the characteristics of a product, for the parameter that is being tested, since the first test may have affected the performance of the product or the intensity of the loads being poorly measured [74].

In the development of a satellite product, the experimental verifications should be done in separated parts [43]. But even with tests, in these separated parts, the construction of a final model is mandatory, at least a prototype model to implement complete experimental tests. There is a possibility of building SM - Structural Models, QM - Qualification Models, FM - Flight Models, among others, for development, qualification and acceptance, or like is done more recently, the SM and the QM is subtracted and replaced by one single model, the PFM - Protoflight Model, which has the same functions as the QM [60].

Figure 2.9 from Ref. [22], represents in a simple way the different phases of tests and can be consulted that the first thing to do is to qualify the main units, the major parts of a satellite. Then the PFM is used to carry out static and dynamic tests with qualification loads (due to the lack of data from previous tests, and to the qualifications loads being the highest loads to be applied in a satellite, the cracks in this model are frequent [108]). The FM is built with considerations from the PFM, facing acceptance loads to prove the integrity of the mechanical system and to discover possible production deviations that were not detected in inspections [60, 74].

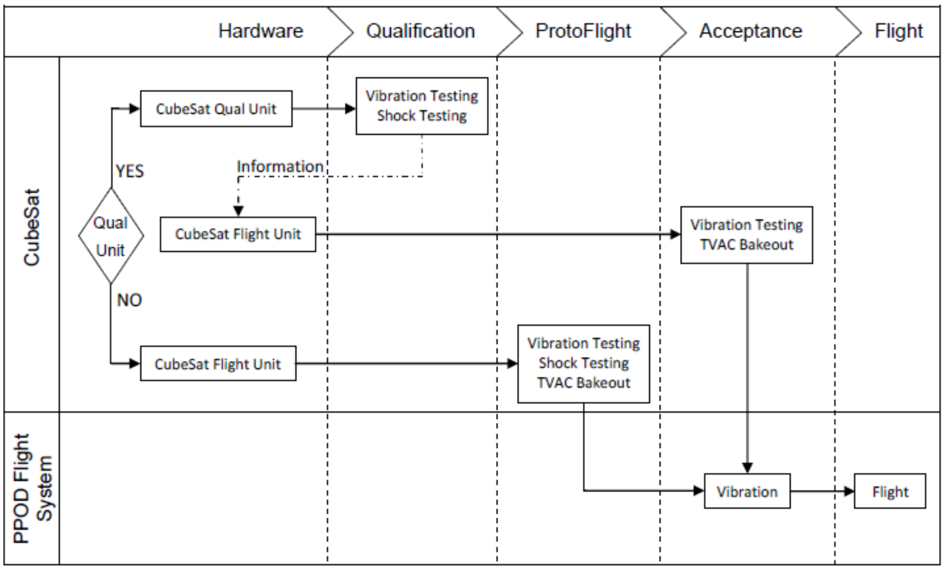

Figure 2.9: Testing flow diagram of a CubeSat with P-POD as the deployer system (figure from Ref. [22]).

When consulting various references, changes are indicated in each type of test to perform in each situation, but some of the tests are always highlighted. A summary identifying the type and the purpose of the most common tests are described below.

Static Load Test

The static load test is performed in order to qualify the primary structure, the connections (joints) and the various internal and external interfaces of a satellite. Alongside with the qualification of structural elements, this test allows finding the main load path [43]. The static experimental verification studies the structure capacity to withstand the quasi-static loads without any permanent deformation or failure, in agreement with the FOS and MoS defined previously. It also gathers relevant information about the stiffness properties of the space vehicle structure, in order to obtain a correlation between the stiffness matrix of the test and the structural finite element model used in the analysis [17].

Modal Survey Test

Aside from the verification of the design frequency, the modal survey test contributes with information for the mathematical model validation using the damping characteristics of the test's items [43]. In the verification of the dynamic behaviour, the natural frequencies, the modal damping, the mode shapes, and the modal effective mass are the aspects to be measured. In the mathematical model validation, the modal survey test is used to correlate the mass and stiffness matrices of structural finite element model and to demonstrate compatibility with the launcher vehicle minimum natural frequencies [94].

Sine Vibration Test

This test is important for the secondary structure definition and qualification. It verifies, just like the modal survey test, the compatibility of the space vehicle with the launcher, the damping properties of the space vehicle and validates specific and really important equipment like antennas or solar arrays [43]. It is also used in the dynamic mathematical model verification, more precisely with correlation with the Sine/Harmonic Response Analysis [43].

Random Vibration Test

The random vibration test is widely used during the development and qualification of spacecraft parts, to be sure that the problems are reduced in advanced stages of the project [42].

Acoustic Test

The acoustic test is really suitable to analyse large and lightweight structures (like solar arrays and antennas), since, these parts are very sensitive to SPL - Sound Pressure Levels [60]. It is also used to qualify the secondary structure when subjected to a dynamic environment and to verify the structural integrity of the space vehicle [43].

Shock Test

The shock test is mandatory to be performed in space systems, in order to evaluate the capacity of the satellites to overcame the launch vehicle separation phases, once the separation phases are the main source of shock loads [108]. The shock loads of high frequency are often responsible for causing failure in electronic components or in mechanisms. The launcher shock tests and the internal shock tests are used to qualify the internal subsystems and to characterise the shock transfer function [17].

Thermal Elastic Test

This test is executed to measure the thermoelastic deformation caused by high temperature gradients, to confirm and validate the numerical model used in the thermo-elastic analysis and to collect data about the parts, boards and assembly gradient temperatures. This data will demonstrate the success or insuccess in the spacecraft operation over a wide range of temperatures [80, 81]. More precise information about the verification process will be discussed in the chapter 3.

The basic vision of the verification and validation process, to be carried out on a satellite was presented in this chapter 2. Thus, it is now the circumstance to present a more concise and direct approach on key subjects of these themes.

Chapter 3

Tailoring of standards for satellite structural development

In the previous chapter, the importance of management in several areas of a project was argued, the ECSS disciplines that have relevance to this dissertation were selected, and an overview of the verification process for small satellites has been made. The goal of this chapter 3 is to go beyond and reveal the most important requirements, recommendations and good practices revealed in the selected standards.

The process of tailoring is never a single, individual and separated selection of condition in a specific area or discipline, but rather a draft, where everything is connected to a large set of similarities between subjects, and a deep understanding is needed in various areas of a satellite development.

In the selection of requirements is necessity to approach important topics like the mission, the functionality, the mechanical interfaces and the design of a spacecraft. Other subjects that should be approached in this theme are the stages, methods and levels of verification; the test methodology, objectives and main procedures to be considered, and the test documentation to be followed, for each project option that is made.

Along with requirements are the test definition, specified in subsection 3.1.3. The test definition covers the model selection, the test sequence or the test tolerances. The different analyses to be performed are reported in subsection 3.2.3, where a set of conditions to perform numerical verifications are presented with considerations about the constraints, procedures, checks and ways to confirm the suitability of the analysis to the case that is intended.

In order to enrich the discussion, after the selection of the main requirements a parallelism between the main specifications from ESA and NASA organization were performed. With it is intended to review both organizations and complete the summary of the most demanding set of requirements outlined by ESA and NASA.

3.1 Systems Engineering

3.1.1 Systems Engineering general requirements

The tailoring process aims after the identification and examination of the project strategies and characteristics, to select the standards and requirements that have to be applied in the program. And that ends up in a set of specifications and guidelines to be used in the development of a product.

One of the tasks of the standard System Engineering general requirements - Ref. [62] is to define and coordinate the Verification Plan. In Appendix C - Documents per Delivery, a reference to all documents that need to be compiled and the corresponding standards are displayed. From it and from Figure 3.1 can be highlighted some important documents besides the VP, or the VCD already described in section 2.3.

Figure 3.1 shows a review of the general requirements, with an approach to the following main subject: engineering requirements; analyses; design and configuration; alongside with verification; and integration and control. The subjects were presented from a management perspective and taking into account the project MECSE application.

The TS - Technical requirements Specification; the DDF - Design Definition File; and the DJF - Design Justification File are the basic documents used in the development of a product. The TS establishes the purpose of a product, associated constraints and environment, operational and performance features for significant events in the life cycle of a product; the DDF is the basic structure that establishes all the information related to the functional, physical architectures and characteristics of a product or system; and the DJF is the document that compiles all the reasons for the selection of a specific design solution, and it also demonstrates that the design meets the baseline requirements. In the DJF other relevant topics are discussed, such as FOS and MoS, the qualification and acceptance criteria among other themes [62].

3.1.2 Verification requirements

Compared to the System engineering general requirements the Verification standard - Ref. [50] has more information about the verification process.

The main themes covered by this standard are: the verification planning, where AITP stands out; the verifications stages, methods and levels already identified in section 2.3; categories of product, sparing the products that can be qualified by similarity and those that can not obtain qualification using this method; the verification execution and reports, which operate with reference to the VP and the DJF; and the verification control and close-out, with the ambition of producing the VCD, which is controlled by the VCB. Useful documents normally added to the AITP and the VP are the Verification Matrix and the Test Matrix.

For an overview of these topics and where they can be found consult Figure 3.2. In comparison NASA organization also indicates some important documentation, that should be compiled, being it enunciated at Ref. [52]. Although the number of documents required to support the VP are smaller at NASA then at ESA, it introduces nearly the same information, since a single NASA document covers a larger number of subjects compared to ECSS standards.

3.1.3 Testing requirements

A significant part of the verification process is carried out by Testing, and therefore a complete standard is dedicated to this topic. The standard Testing - Ref. [74] defines all the test management requirements, along with the tests levels and durations that must be performed at equipment and element level. In Figure 3.3 documents are presented to be completed before, during and after every experimental verification.

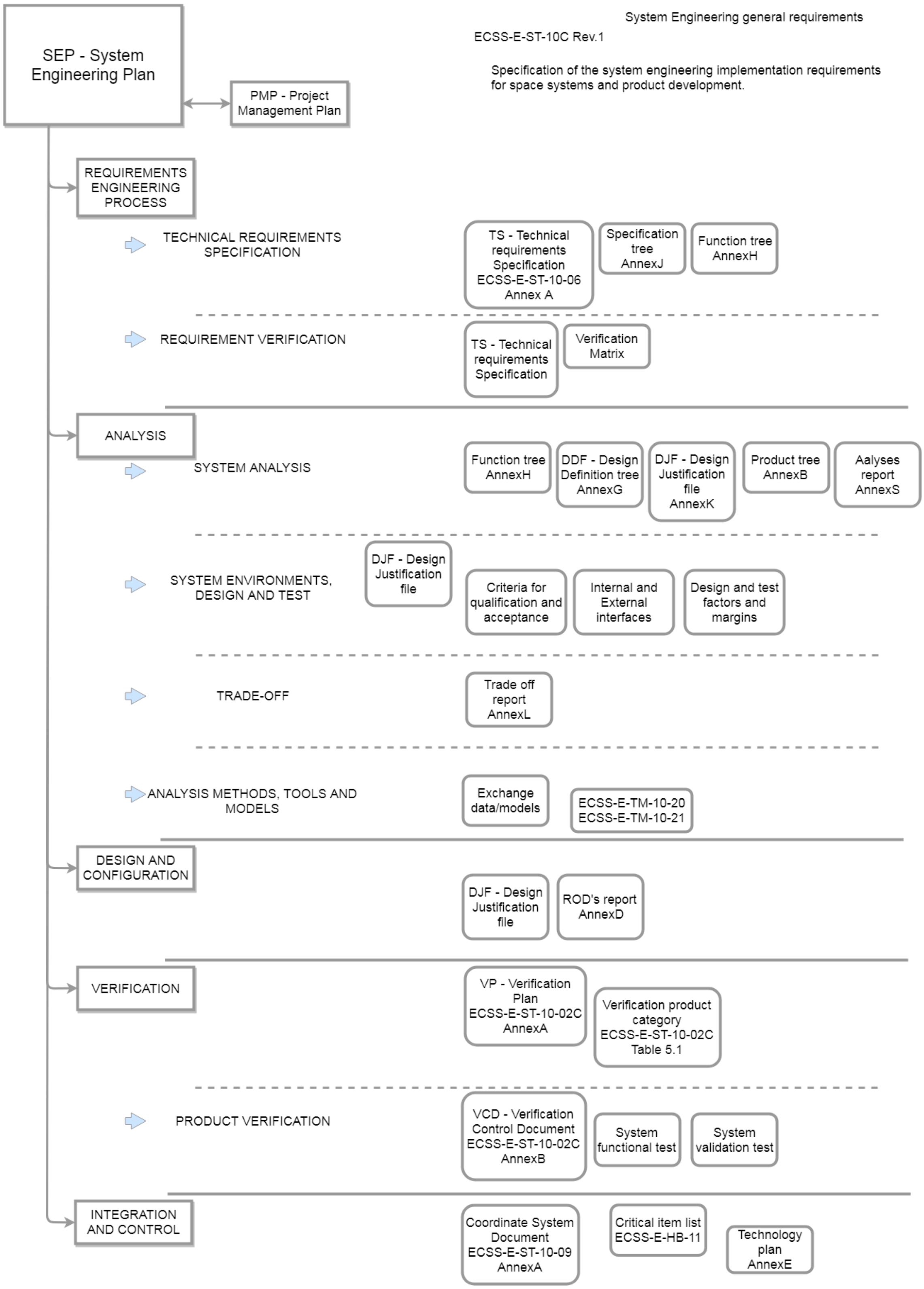

Figure 3.1: System general requirements documents (figure compiled in agreement with Ref. [62]).

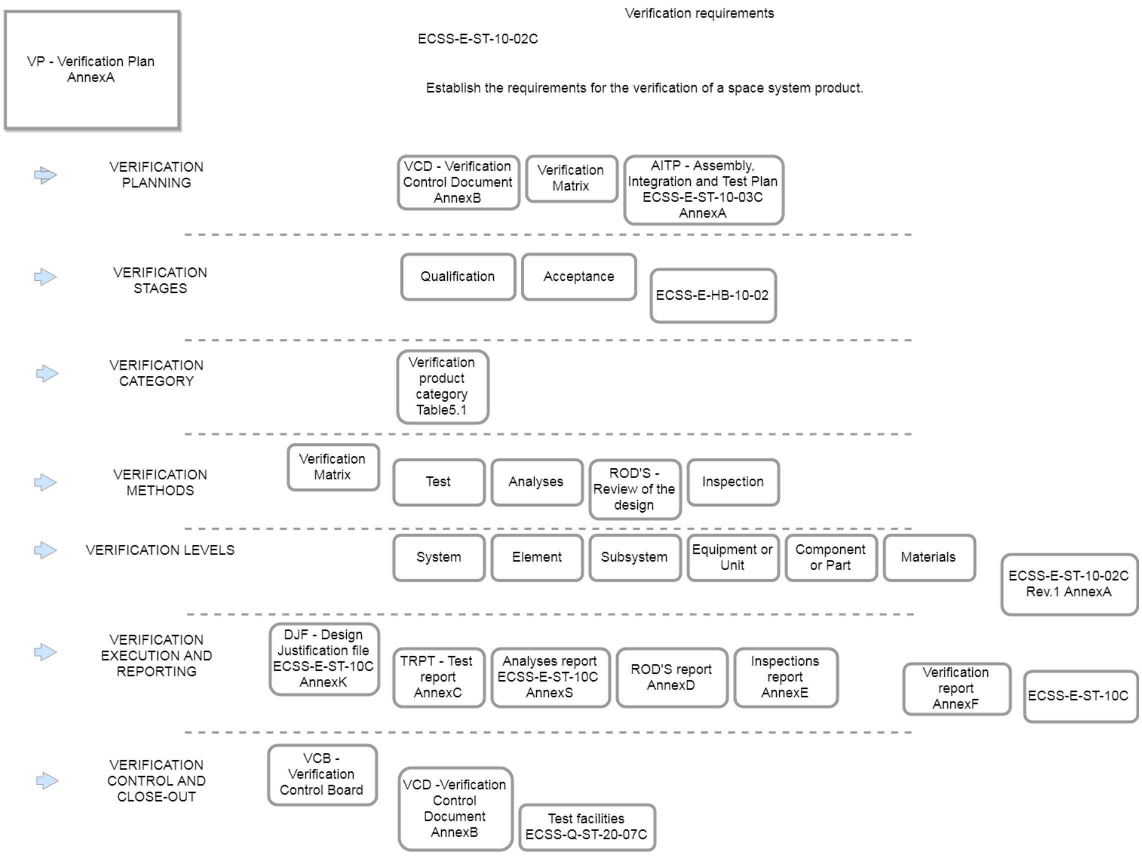

Figure 3.2: Verifications requirements (figure compiled in agreement with [50]).

The main documents that should be considered when planning a test is a task to be performed are: the TSPE - Test Specifications of a product, which are constructed with information about the product requirements; the VP and the VCD. Equally important for the organization of the experimental verification are the TPRO - Test Procedures, formulated according to the TSPE and the AITP; the test blocks, that compiled formulate the complete test programme, and include the formal reviews TRR - Test Readiness Review[1], and the PTR - Post Test Review both controlled by the TRB - Test Review Board. The TRPT - Test Reports indicates the test conditions, test tolerances and accuracies and it is an important part of the verification program. The model philosophy that is used in experimental verification is another subject to be described.

Defining the model philosophy is a crucial part of the overall planning of verification. It will define part of the project's development and it will define the analyses and test conditions. The requirements to apply in analyses and test cases should be the same, in order to allow correlations between the results presented in the numerical analysis and the experimental test that will occur.

The model philosophy can be distinguished between prototype philosophy, protoflight model philosophy or hybrid philosophy. The decision to use which philosophy is a balance between the cost, the schedule, the risk to be taken in the program, among others parameters.

[1] TRR - Test Readiness Review is an important review as it checks the pre-test conditions and it allows the test programme to continue. The TRR should approach the following topics: test documentation, with the AITP, TSPE, test predictions, TPRO, measurement point plans and test facility readiness report; configuration of test items; tests set-up; inspections reports; waivers status and deviations; tests pass and fail criteria; and to complete, the alignment of responsibilities [74].

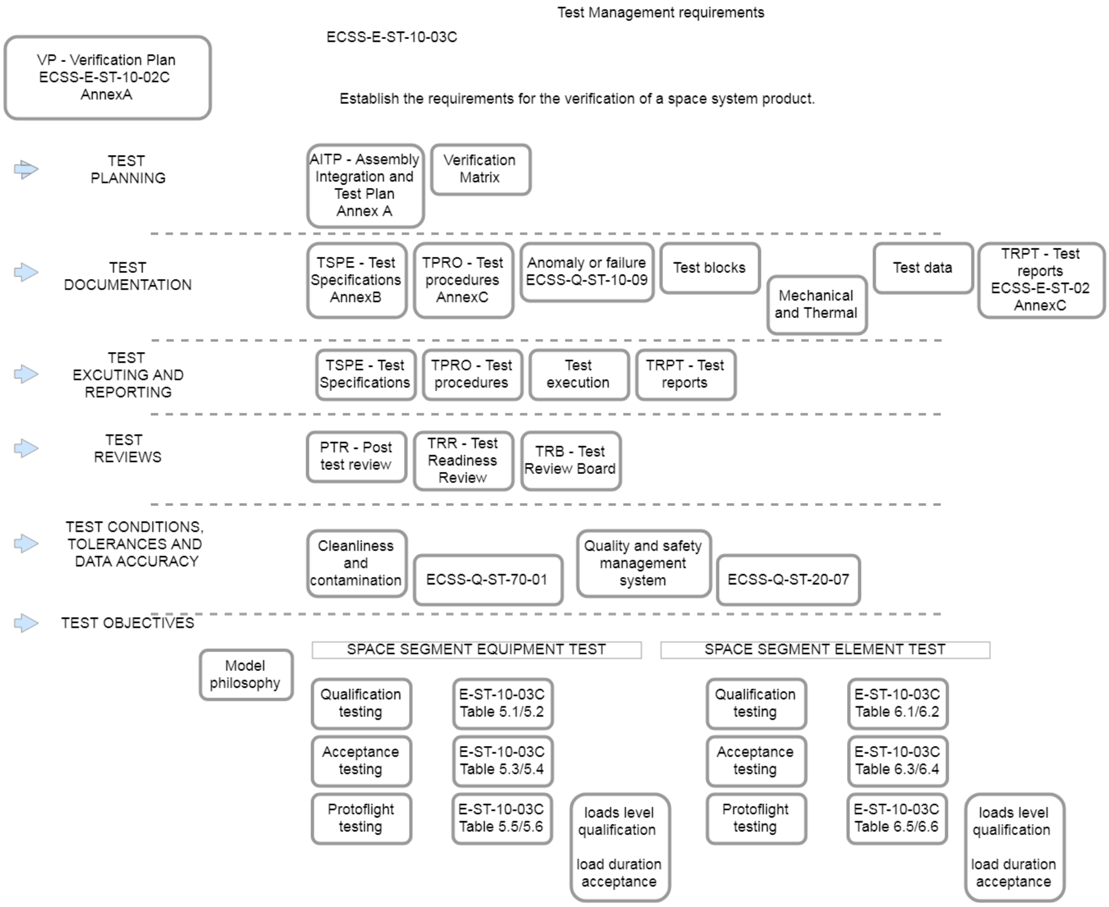

Figure 3.3: Test Management requirements (figure compiled in agreement with [74]).

3.1.3.1 Model philosophy selection

After examining the various condition presented in Figure 3.4, and using Refs. [60, 61] as baseline support, with information on the models to be built, the tests that need to be performed and the type of test on each part of the satellite that needs to be run, an evaluation among the available possibilities will be granted.

The protoflight philosophy model distinguishes the satellites hardware qualification between equipment and complete system (in comparison the prototype model performs verifications in three levels of the spacecraft; equipment, subsystems and systems, and the hybrid approach still adds the payload instruments verification level to the prototype model levels).

The prototype model approach is the model philosophy with more costs associated with its use since, it is applied in satellites with proposals for a completely new design and configuration, not appropriate conditions to be employed in MECSE project.

The PFM is used in projects with similar characterised to MECSE, once it is a university project, with a unique scientific mission, a limited budget, with most of the equipment and products widely used in space applications (information according to Ref. [47]), which would lead to the protoflight approach a suitable response to this situation.

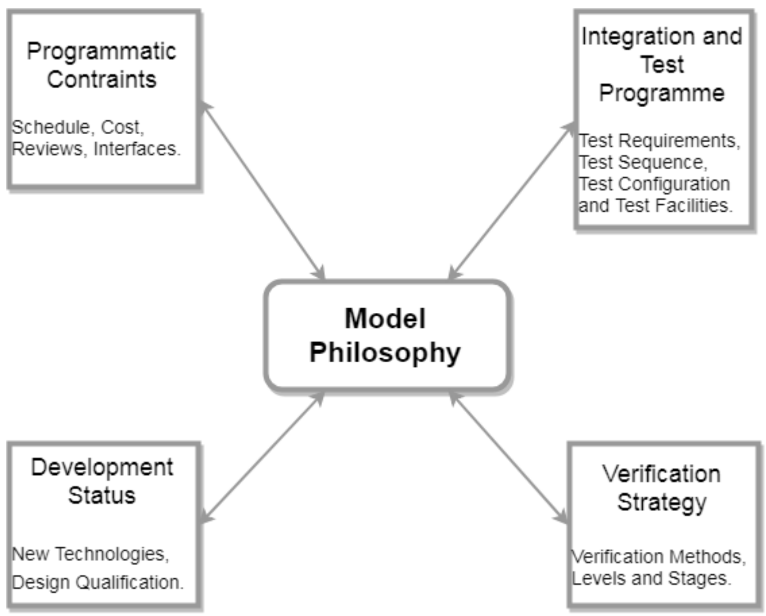

Figure 3.4: Parameters to choose the model phylosophy (in agreement with Ref. [75]).

Associate with the PFM is the creation of a single model, and therefore some understandable drawbacks are inherent, such as an increased risk, mixed qualification and acceptance activities and a lack of model spare parts are the main ones [75].

Thus, the primary mission objective of this project is to provide practical experience to students on space projects, and because of it is intended to build a STM - Structural-Thermal Model. This model will be a representative structure at system level, equipped with flight equipment dummies, to predict structure response (anticipate deformations, natural frequencies, internal forces and stresses). Helping in the design of an optimized, realistic and appropriate CubeSat structure. It will also check the effectiveness of the payload accommodation, and in the end, it will give the responsible engineer the opportunity to learn about the experimental verifications on an initial model and not on the final model. For those reasons, a hybrid approach was chosen.

With this hybrid model philosophy is intended to include in the verification strategy virtual models (numerical models), a STM, an EQM/EM - Engineering Qualification Model / Engineering Model and a physical model with the protoflight attributes (the STM can be refurbished and presented to be the PFM, reducing the programme cost).

Decoupling the STM activities, from the EQM/EM activities enable the verification flow of the satellite project to continue, avoiding an interruption due to hardware unavailability (which reduces schedules risks in the programme). In addition, greater confidence in the project development is granted compared to the protoflight model philosophy, once better conditions for the project development are achieved. The equipment in the EQM/EM remains to be qualified at equipment level according to the available units [75]. Apart from that reasons the main advantages of performing qualification in lower levels of assembly (equipment level and sub-system level) is the amplification of time to diagnose and fix problems founded in these tests, avoiding further problems on the satellite [109].

The Figure 3.5 shows an example of a hybrid model, which can be applied to MECSE. The various types of equipment are qualified according to the EQM instructions, and the EM (with the functional qualification of the project).

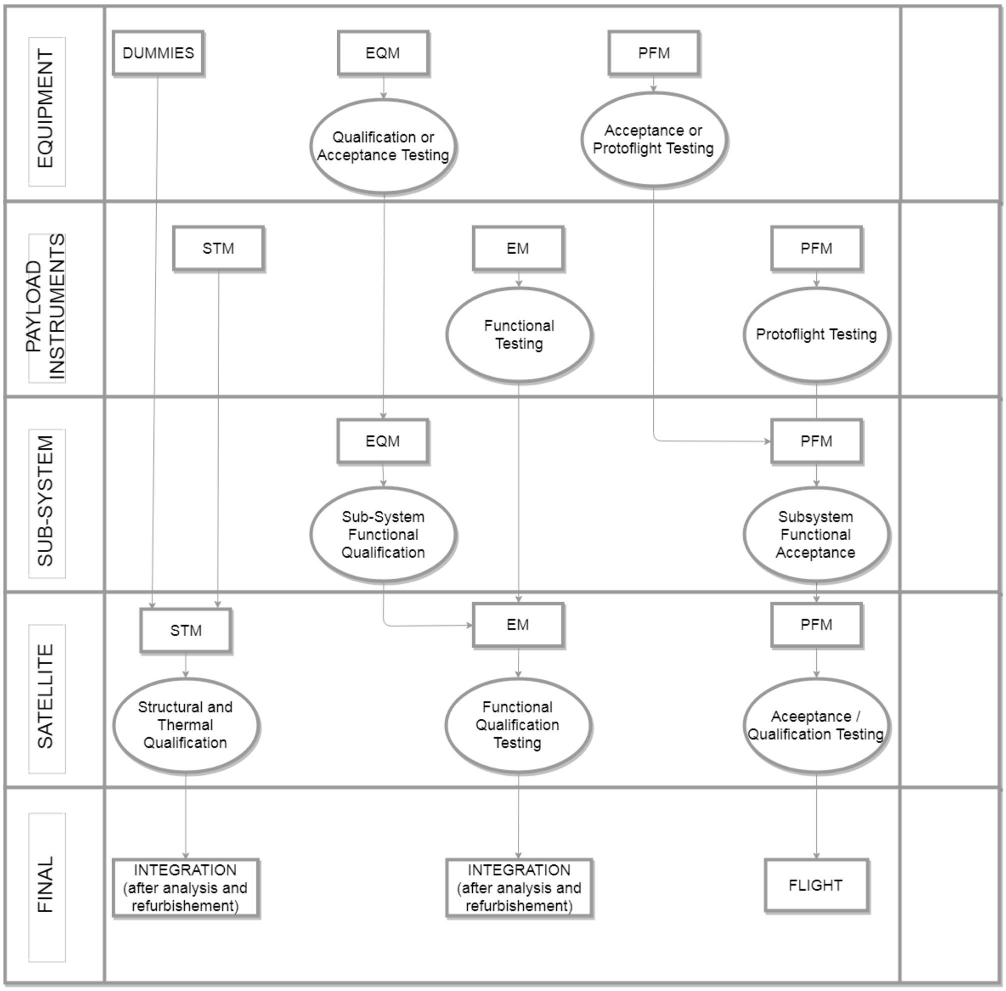

Figure 3.5: Hybrid model phylosophy (adapted from Refs. [61, 75] in agreement with MECSE project structural demanding).

The EQM/EM is used to study the functional qualification of the equipment and to collect information about functional deviations that can be confirmed with these tests. The PFM which will reproduce the FM functions is tested at all the system levels. To note that, the PFM cannot suffer any modification, or when modified it can only be submitted to qualification tests once to prevent overtesting [61].

At equipment level the notching technique is often used to avoid overtesting (at system level, overtesting is less likely to happen and because of this, it is not so common to use the notching [109]). This is a technique that must be accepted by the launch vehicle authority, and it is achieved with a reduction on the excitation levels of the sine and random vibration tests, culminating in a more efficient management of the load's intensity, especially at structure interfaces, where the most intense loads occurs [109].

3.1.3.2 Test sequence

The complete test sequence system can be seen in Figure 3.6, where the tests are presented according to ESA documentation. In standard Testing Ref. [74], the test sequence is well-defined, relative to the model philosophy chosen, and to the specific case of satellites.

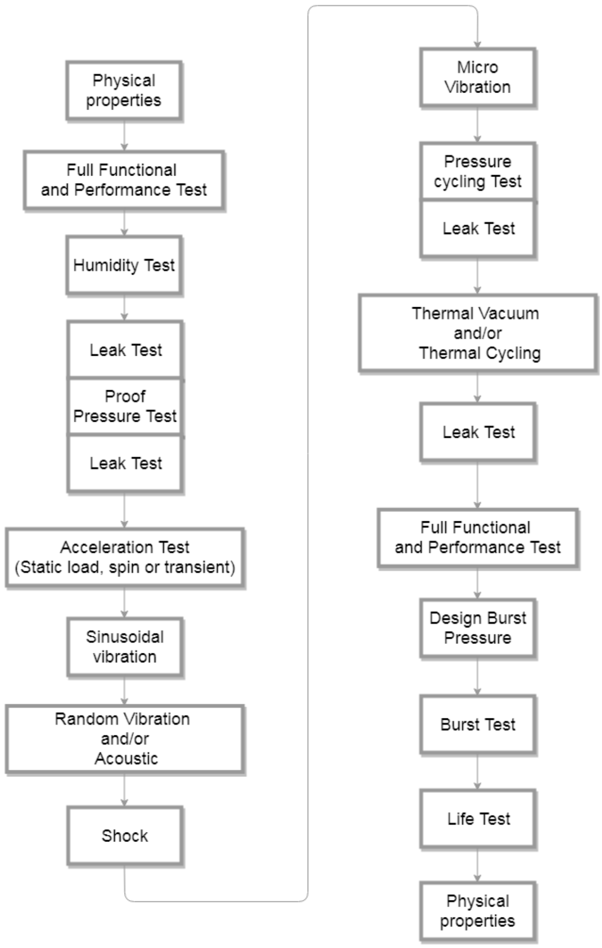

Figure 3.6: Space segment test sequence (figure compiled in agreement with Ref. [74]).

NASA does not have such a strict test sequence, although it does suggest that the sequence should be carried out according to the launch environment sequence, which represents the test sequence presented in Figure 3.6. This flexibility, in the test sequence leaves to the responsible engineers the definition of the test sequence, and in some cases this option assists in the development of the project due to the possibility of performing the following tests if the current test lacks of materials to be performed, or if the test condition is not granted for the actual test (consulting Ref. [110] this theme is explained in more detail).

As the aim of this work is to focus on the structural subsystem and evaluate the different responses of a CubeSat structure, the test sequence to be applied in MECSE must be followed without deviations from the previously presented. And a selection between the required tests and the optional tests will be presented, specifying the test, test levels and durations to apply at equipment and element level.

3.1.3.3 Equipment test

Equipment level testing is intended to assure the integrity of the element and evaluate whether the connections between parts can withstand the loads by which these items will be subjected, being this the main focus of the micro-vibration test. In addition to these reasons the test of an element has the purpose of revealing possible errors in an early phase of the project and to qualify these units, as presented earlier.

Among space segment equipment, some are the types of equipment that are critical to a small satellite structure and these will be the equipment to suffer experimental verification. The critical list of equipment is chosen in accordance with their function, or due to their substantial percentage of mass compared to the spacecraft's global mass [76]. In agreement with ESA, the following items were selected to be part of the critical list: batteries; pressure vessel; thruster and thermal equipment. Even though MECSE has not planned to adopt these last three types of equipment, they are present in Figure 3.7 for an overview of the critical items and the equipment tests to perform on a CubeSat. To outline that, if in Figure 3.7 any indication is given, the test must be performed on every equipment in the critical list.

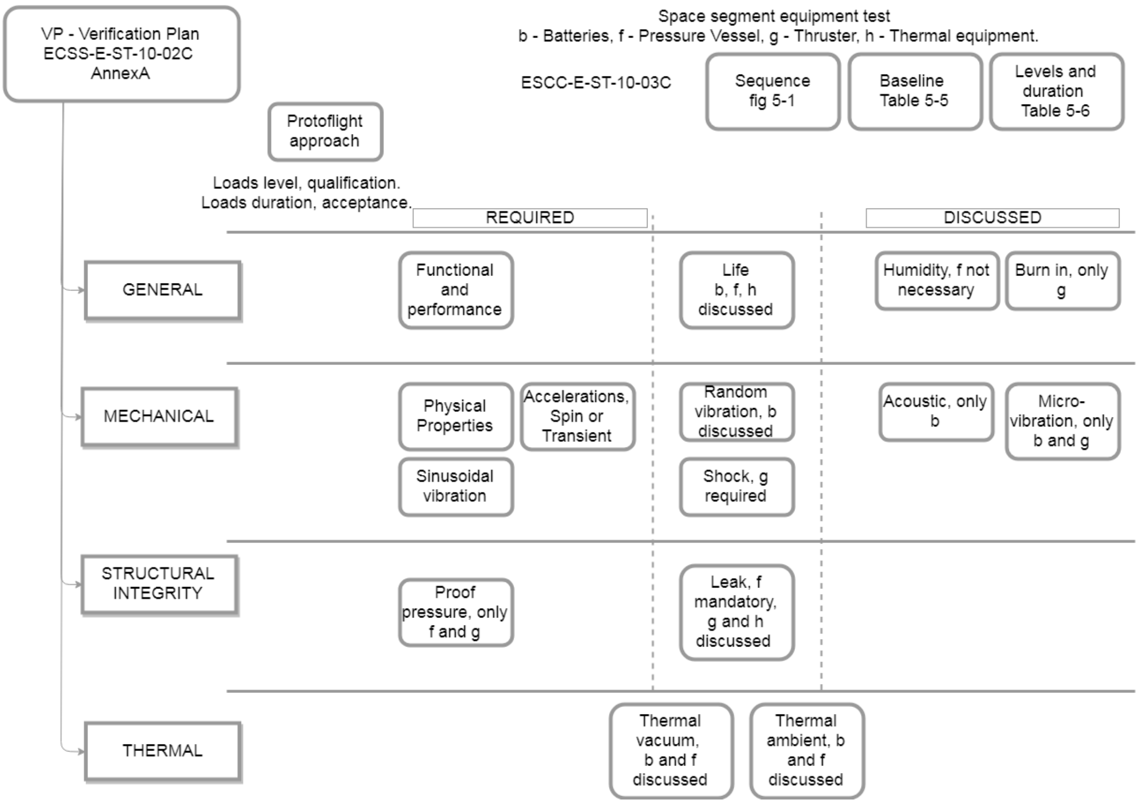

Figure 3.7: Space segment equipment, protoflight test baseline (figure compiled in agreement with "Table 5-5: Space segment equipement - Protoflight test baseline" from [74]).

The test levels and the test durations to be applied in critical items are specified in the Testing standard. NASA introduces test levels and test durations for equipment in Ref. [52] and Ref. [111]. Due to distinctions between ESA and NASA documents, an overview of the test levels and durations was formulated and presented in Table 3.1 with the worst case among the data collected from the documents listed above.

Table 3.1: Space Segment, equipment protoflight test levels and durations (figure compiled in concordance with Ref. [52], Ref. [74] and Ref. [111]).

Test	Level	Duration	Number of applications	Notes
Life	Expected environment and maximum operational loads	For batteries, apply ECSS-ST-20	1 test	
Spin	$\sqrt{KQ}$ x Spin rate	As needed to record data	On each of the 3 axes	
Transient	KQ x Limit load or Limit load Spectrum + 3 dB	As needed to record data	2 times for each of the 3 axes	
Random Vibration	Maximum expected spectrum + 3 dB PSD value	3 min	On each of the 3 axes	
Acoustic	Maximum expected acoustic spectrum + 3 dB	3 min	1 test	
Sinusoidal vibration	KQ x Limit load Spectrum	Sweep at 4 Oct/mim 5 Hz to 140 Hz	On each of the 3 axes	
Shock	Maximum expected shock spectrum + 3 dB	typical between 20 ms and 30 ms	1 test	If the test is performed in a shock generative device, the margin is not necessary.
Microvibration susceptibility	Maximum predicted environment	As needed for susceptability determination	As specified by the project	
Leak	MDP - Maximum Design Pressure	Pressure maintained for 30 min as minimum	As specified in	
Proof pressure	j_{proof} x MDP	5 min minimum hold time	1 test	
Thermal Vaccum	$T_{max}^{Q\ or\ A}\|_{OP\ or\ NOp} = +/-5\ C$ Applied for temperatures between - 170ºC and 120 ºC Bigger margins can be applied for temperatures out of this range	4 cycles For solar panels, 10 cycles	1 test	The Thermal tests are both performed for space segment equipment that operate under a non-vacuum environment after having been exposed to vacuum.
Thermal Ambient	$T_{max}^{Q\ or\ A}\|_{OP\ or\ NOp} = +/-5\ C$ Applied for temperatures between - 170ºC and 120 ºC Bigger margins can be applied for temperatures out of this range	4 cycles or 4 cycles minus the number ofcycles performed in the vacuum test	1 test	This test without the vacuum test is applicable only to space segment equipment that operate under a non-vacuum environment during their entire lifetime.

3.1.3.4 Element test

As enunciated in subsubsection 3.1.3.1 and Figure 3.5 the model methodology assumes not only the equipment qualification though tests, but also the acceptance and qualification tests of the protoflight model. Following the selected standard and the purpose of this work, the tailored element tests are present below, in Figure 3.8.

The PTF should be tested following the protoflight levels and durations which are characterised by the qualification load limits levels, and with the duration of the load acceptance. In Table 3.2 all the tests and their applicability in agreement with Ref. [74] are presented. It is also important to refer that in both tests (equipment and element) the space segment equipment and element must be tested in launch configuration through its normal mounting points.

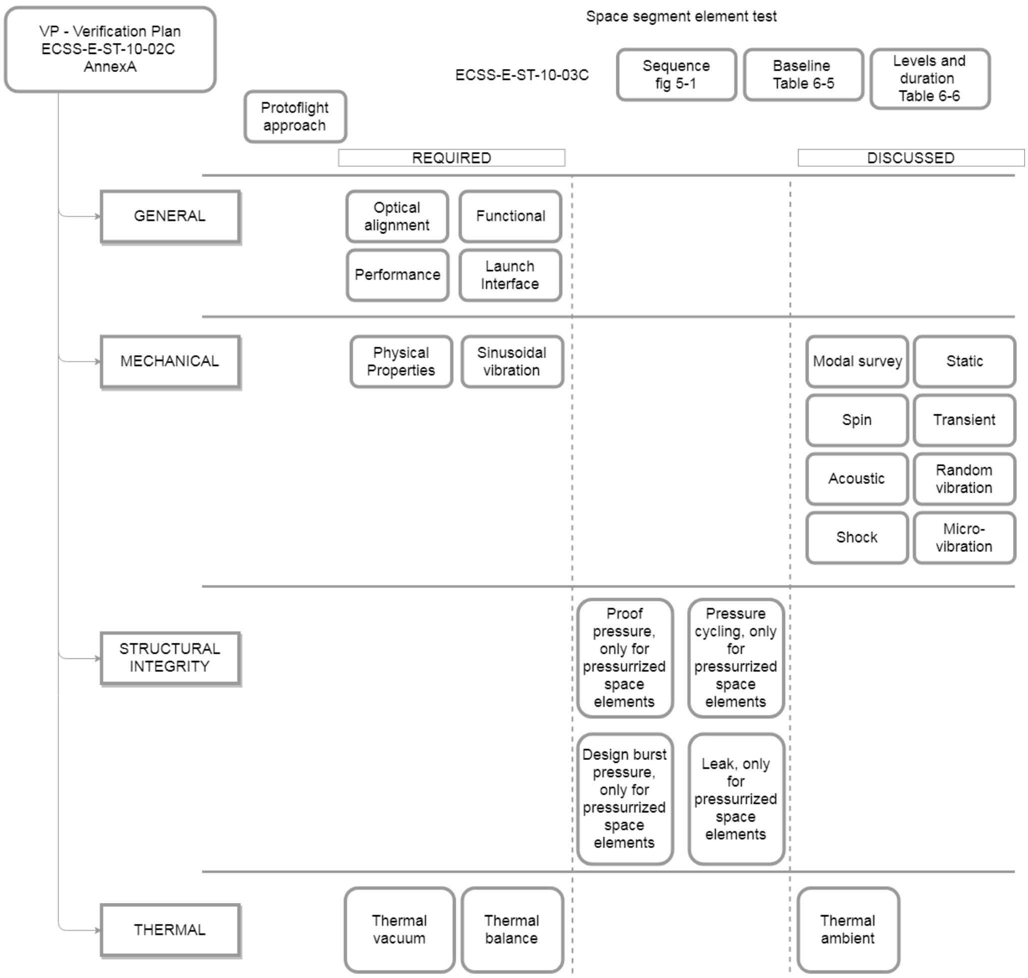

Figure 3.8: Space segment element, protoflight test baseline (figure compiled in agreement with "Table 6-5: Space segment element - Protoflight test baseline" from Ref. [74]).

The environment tests conditions and specifications defined by NASA organization for CubeSat are delineated in Ref. [68]. This information is completed with two reference documents Refs. [112,113] and it will be discussed in more detail further in this chapter 3 once, these two documents contain information that was not explained in detail (mainly the FOS and MoS), but it will be with the progress into subsection 3.2.1. Being the comparison between ESA and NASA organization about these matters discussed at that point.

Table 3.2: Space Segment, element protoflight test levels and durations (figure compiled in concordance with Ref. [52], Ref. [74] and Ref. [111]).

Test	Level	Duration	Number of applications	Notes
Static Load	KQ x Limit load or Limit load Spectrum + 3 dB	As needed to record data	Worst combined load cases	Determined by analysis
Spin	$\sqrt{KQ}$ x Spin rate	As needed to record data	On each of the 3 axes	
Transient	KQ x Limit load + 3 dB	As needed to record the test data	2 times for each of the 3 axes	
Random Vibration	Maximum expected spectrum + 3dB PSD value	2 min	On each of the 3 axes	
Acoustic	Maximum expected acoustic spectrum + 3dB	2 min	1 test	
Sinusoidal vibration	KQ x Limit load spectrum	Sweep at 4 Oct/mim 5 Hz to 100 Hz	On each of the 3 axes	
Shock	Limited to the test whit shock generative device is actived. No margins of safety are necessary to consolidate the sock.	Representative of the expected environment	1 activation	If the test is performed in a shock generative device, the margin is not necessary.
Microvibration susceptibility	Maximum predicted environment	As needed for susceptability determination	As specified by the project	
Leak	MDP - Maximum Design Pressure	Pressure maintained for 30 min as minimum		
Proof Pressure	j_{proof} x MDP	5 min minimum hold time	1 test	
Design burst pressure	j_{brust} x MDP	30 sec as minimum	1 test	
Thermal Ambient	Applied to the maximum and minimum temperature prediction +/- 10 ºC	3 cycles or 3 cycles minus the number ofcycles performed in the vacuum test	1 test	This test without the vacuum test is applicable only to space segment equipment that operate under a non-vacuum environment during their entire lifetime.
Thermal Vacuum	Applied to the maximum and minimum temperature prediction +/- 10 ºC	3 cycles + 1 back up to be decided during tests or 1 or more cycles if combined with ambient	1 test	The Thermal tests are both performed for space segment equipment that operate under a non-vacuum environment after having been exposed to vacuum.

3.1.3.5 Test Tolerance

The tests errors associated with the generation of the loads and their measurements are the main cause of uncertainty in the experimental verification. This leads to the need to set the test tolerances and test accuracies. The allowed test tolerance to take place in the introduction of the loads into an experimental verification is defined in Ref. [74] and by NASA it is given in Ref. [52].

In this particular case, the worst scenario between ESA and NASA changes in agreement with the parameter to be studied. Because of it, Table 3.3 has been constructed representing an overview of the deviations from the most demanding test parameters that must be taken into account in an experimental verification.

Table 3.3: Test Tolerance table (compiled with information from Ref. [52] and Ref. [74]).

Test Parameter		Tolerance
Temperature		
	above 80k	Tmin = +0 / -2 K Tmax = +0 / -2 K
	below 80k	Defined case by case.
Relative Humidity		+/- 5 %
Pressure (in vacuum chamber)		
	>13 000 Pa	+/- 5%
	from 130 Pa to 13 000 Pa	+/- 10 %
	from 13 Pa to 130 Pa	+/- 25 %
	<13 Pa	+/- 80 %
Acceleration (steady state) and static load		+5 / -0 %
Sinusoidal vibration		
	Frequency	+/- 2 % (or +/- 1 Hz, whichever is greater)
	Amplitude	+/- 10 %
	Sweep rate (Oct / min)	+/- 5 %
Random vibration		
	Amplitude (PSD, frequency resolution better than 10 Hz)	
	from 20 Hz to 1 000 Hz	+3 / - 1 dB
	from 1 0000 Hz to 2 000 Hz	+/- 3 dB
	Random overall, RMS level	+/- 10%
	Duration	+ 10 / - 0%
Acoustic noise		
	Sound pressure level, Octave band center (Hz)	
	f = 31.5 Hz	+3 / - 2 db
	from 40 Hz to 3150 Hz	+3 / -1 dB
	f = 3150 Hz	+3 / -6 dB
	Duration	+10 / -0 %
Microvibration		
	Acceleration	+10 / -0 %
	Force and/or Torque	+10 / -0 %
Shock		
	Response Spectrum Amplitude (1/12 octave centre frequency or higher)	
	Simulated	+6 / -3 dB
	Shaker	+/- 3 dB
		>50 % of the SRS amplitude above nominal test level (0 dB)
	Time history	+10 / -0 %
Mass properties		
	Weight	+/- 2%
	Center of gravity	+/- 0.15 cm
	Moments of Inertia	+/- 1.5 %

3.1.3.6 Test Accuracy

The test tolerance has been defined, but also the deviations from data collection need to be specified. The tests errors associated with the measurements on tests instruments are related to the instruments calibration and the allowed deviations are presented in Table 3.4, with information compiled from Ref. [74].

Table 3.4: Test Accuracy table (compiled with information from Ref. [74]).

Test parameters		Accuracy
Mass properties		
	Space equipment and element	The heavier between +/- 0.05 % or 1g
	Center of gravity for equipment	1 mm radius sphere
	Center of gravity for element	+/- 2.5 mm along launch axis +/- 1 along other axes
	Moment of inertia	+/- 3 % for each axis
Leak rate		One magnitude lower than the system specification, in Pa m^3 s-1 at standard conditions (1013.25 Pa and 288.15 K)
Temperature		
	above 80K	+/- 2 K
	below 80K	Defined case by case
Pressure		
	above 130 Pa	+/- 15 %
	from 130 Pa to 0.13 Pa	+/- 30 %
	below 0.13 Pa	+/- 80 %
Accelerations (steady state) and static load		10 %
Frequency		+/- 2 % (or +/- 1 Hz whichever is greater)
Acoustic noise		+/- 0.1 dB
Strain		+/- 10 %

Going through a structure analysis there are some parts more likely to fail than others, there are critical points on the structure depending on the type of load applied. In that perspective NASA states in Ref. [53] which are the most propitious parts to failure and the reason for this failure, just as ECSS does with the Engineering branch standard Fracture control Ref. [114], and the Materials standard Ref. [88].

After indicating the main requirements to be utilized in the system engineering discipline, it is time to present the Mechanical engineering discipline and particularly the Thermal control requirements in subsection 3.2.1 and the Structural requirements in subsection 3.2.2.

3.2 Mechanical Engineering

3.2.1 Thermal-control requirements

For a precise analysis of the thermal conditions that a satellite will face and a correct correlation to the temperature range of operation the Thermal Control general requirements standard, Ref. [79], was taken into account. The temperature of a satellite suffer a significant thermal variation due to the many mission phases that have to be considered in the temperature analysis - launch, transfer orbit, apogee and perigee, eclipse and sunlight modes are the phases to be studied [80].

Considering the standard Thermal control general requirements was possible to signalize the conditions that a satellite will meet and also the analysis and associated tests and documentation that support the verification process of the thermal control subsystem. A summary of it is presented in Figure 3.9.

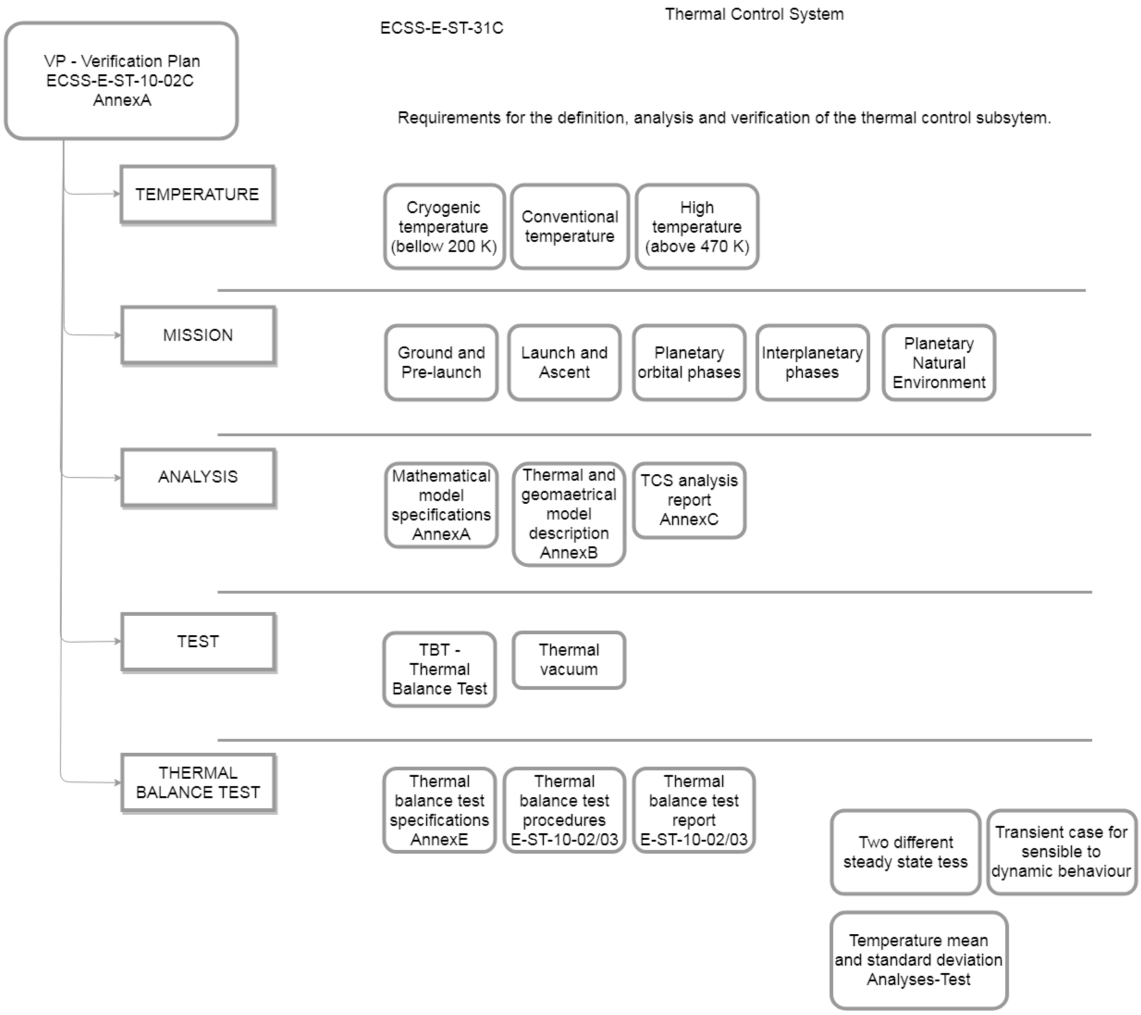

Figure 3.9: Thermal-control system requirements (figure compiled in agreement with with [79]).

3.2.2 Structural General requirements

When it is intended to construct a spaceflight structure, the design requirements identified in section 2.3 have to be followed. Within the intention of a better identification and separation of the vast number of structural requirements, to have in consideration in the different phases of the structure development, a categorization in agreement with Ref. [82] has been made.

Mission requirements from this category may be withdrawn, the Lifetime of the satellite that is always in accordance with the Environment and the Limit Loads that needed to be supported by the satellite structure in the different stages of the life cycle of the product;

Functionality requirements these requirements are used to confirm the performing capabilities of the design, alongside with the monitoring of manufacturing, verification and operation of the hardware units, and it is always a distinction between metallic or non-metallic parts. It intends to accomplish the identification of Strength, Stiffness, Damage Tolerance, Local yielding, Dynamic behaviour, Thermal, Buckling and others characteristics of each part [114];

Interface requirements this class of specifications, evaluate the Internal interfaces between sub-systems and the External interface by presenting the external connections with the launch vehicle. At external interactions is necessary to identify, the needed qualification to provide to the operators with the capacities to handle properly the necessary equipment. Alongside with the preparation of the suitable GSE - Ground Support Equipment, for pre-flight operations;

Design requirements approaches themes like the Inspectability, the Maintainability, the Dismountability, the Material selection, the allowable Mechanical design, the Factors of safety or the Margins of safety and will be discussed specifically in subsection 3.2.4;

Verifications requirements were described in detail in subsection 3.1.2 and subsection 3.1.3;

Deliverables requirements a complete list of the structural deliverables is present in subsection C.1.1, being these documents baselines for some documents requested in the system engineering discipline and the verification process (such as correlation with some of the documents reported in subsection 3.1.1).

The Figure 3.10 presents the refereed categories above and what which one is supposed to have into consideration. A review of the different requirements are presented forward in chapter 4 and in chapter 5.

3.2.3 Structural Finite Element Models requirements

The norms and requirements for experimental verification were already been approached in subsection 3.1.3. Leaving a lack of information in accordance with the numerical verification. The standard Structural FEM, Ref. [84] introduces some concepts and requirements like can be seen in Figure 3.11, but important subjects and notions were left behind and because of it an introduction to analyses is made.

In any engineering analysis, before defining the type of analysis, is mandatory to define the method of analysis between the analytical or classical methods[2] and the numerical approach [115]. The most common numerical method used in engineering is the FEA - Finite Element Analysis, once offers a problem solution based on the simplification of a complex geometry and approximates the behaviour of the structure for the loads that are expected to charge the required body, and it is the one adopted to be taken into consideration for this dissertation [100, 116].

[2] Attempt to solve real problems by formulating differential equations based on fundamental principles, laws and theories of physics, providing solutions, being only possible to be applied in the simplest cases [115].

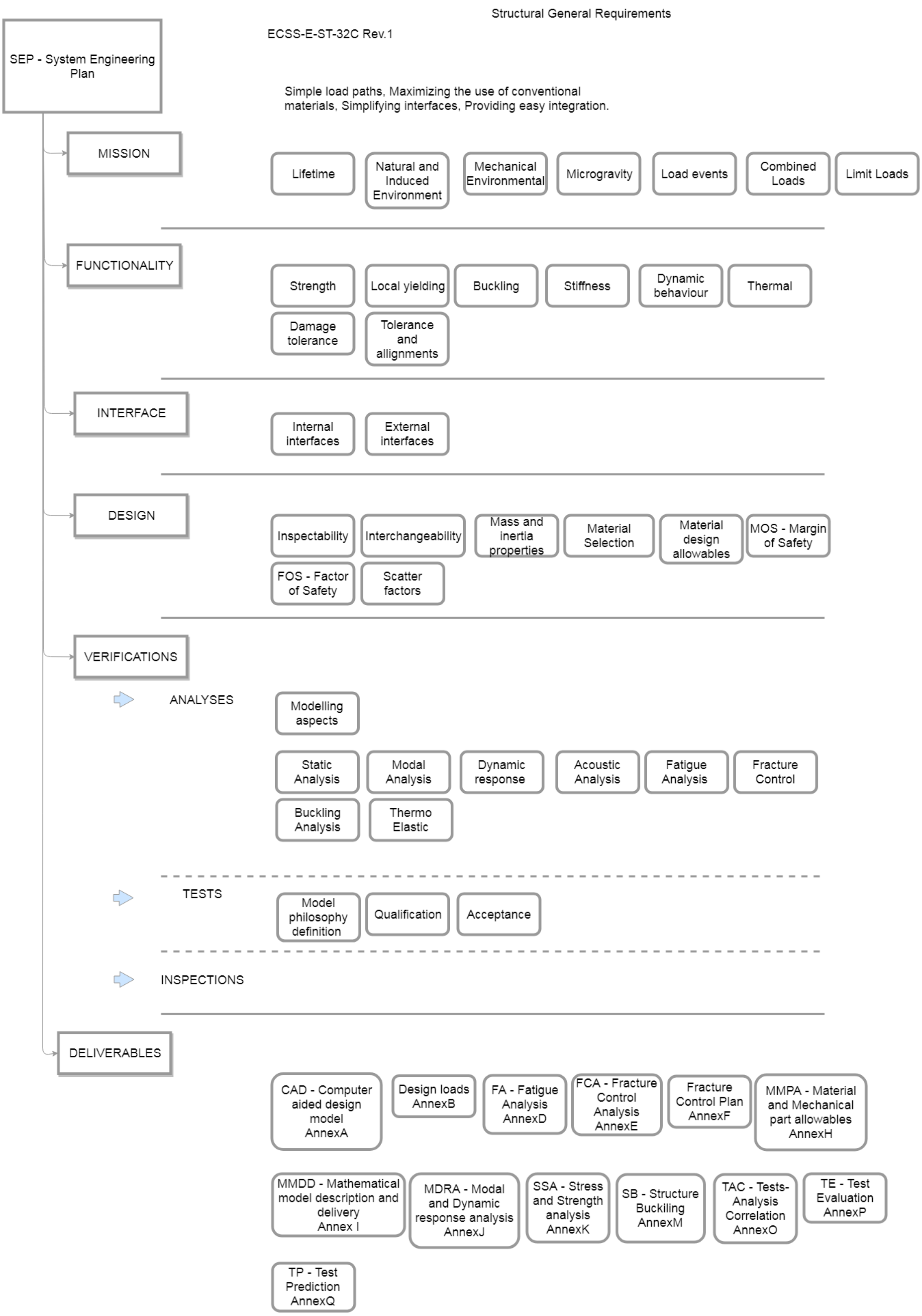

Figure 3.10: Structural general requirements (figure compiled in agreement with [82]).

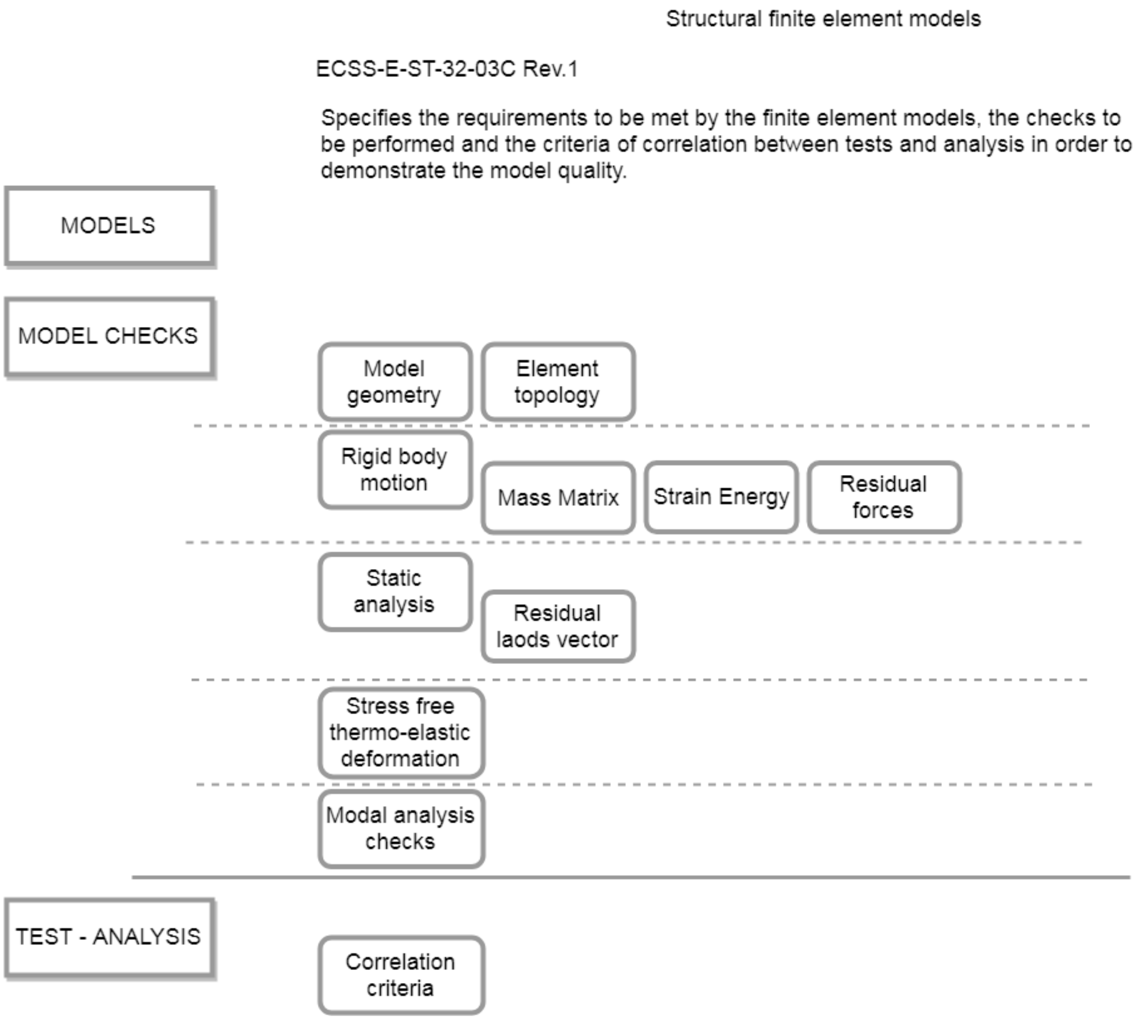

Figure 3.11: Structural FEM requirement (figure compiled in agreement with [84]).

3.2.3.1 Finite Elements Analysis

FEA - Finite Element Analysis or FEM - Finite Element Method (like is called in academic ambient) is used to obtain an approximate solution of a problem.

When FEM is implemented, a large variety of configurations can be given to a problem. FEM allows the discreterization of a domain in smaller elements of various shapes that are interconnected by nodes. The advantage of this method is that, it allows the evaluation of the behaviour of each node, and with that, obtain the approximate solution to the introduced problem [117].

The finite elements fall into four classes: the non-dimensional elements, such as the concentrated masses; the one-dimensional elements, such as the beams or springs; the two-dimensional elements, such as the triangular and quadrilateral surface elements; and three-dimensional elements, such as the bricks or the pyramids solid elements [117, 118].

The main goal of FEM is to determine the unknowns, the degrees of freedom presented in the mathematical formulation of the problem. From these unknows it can be compiled other values in a post-processing step, for example, in the case of a structural analysis, the unknowns are the displacements at the nodes and the stresses are compiled from these [119].

To perform an analysis three generic phases are necessary [100, 120].

(I) Pre-Processing Phase: involves the discretization of the geometry domain into specific finite elements defined by nodes and their connections, specifying the material properties, applying the boundary conditions and the corresponding loads;

(II) Processing/Solution Phase: consists in solving a set of algebraic equations which were previously compiled from the differential equations;

(III) Post-Processing Phase: is the final phase and it is where the results are calculated, visualized, verified and validated in order to confirm their framework. It is also the stage where the results are treated, in order to be shown in a user-friendly and appealing way.

A correct understanding of the numerical approach, built to analyse a problem with FEM, can be achieved with the Figure 3.12.

Structural FEM nodes has six DOF - Degrees of Freedom, six components of motion, the translations and rotations in the three perpendicular directions (e.g., the X, Y, Z, R_x, R_y, and R_z directions).

Equation 3.1, represents the basic numerical model that the solver needs to solve in order to obtain the degrees of freedom in a static structural analysis. The loads applied to the model will be combined to form the load vector. On the other hand, the Equation 3.2 presents the equation that has to be considered for dynamic structural analyses [102, 121, 122].

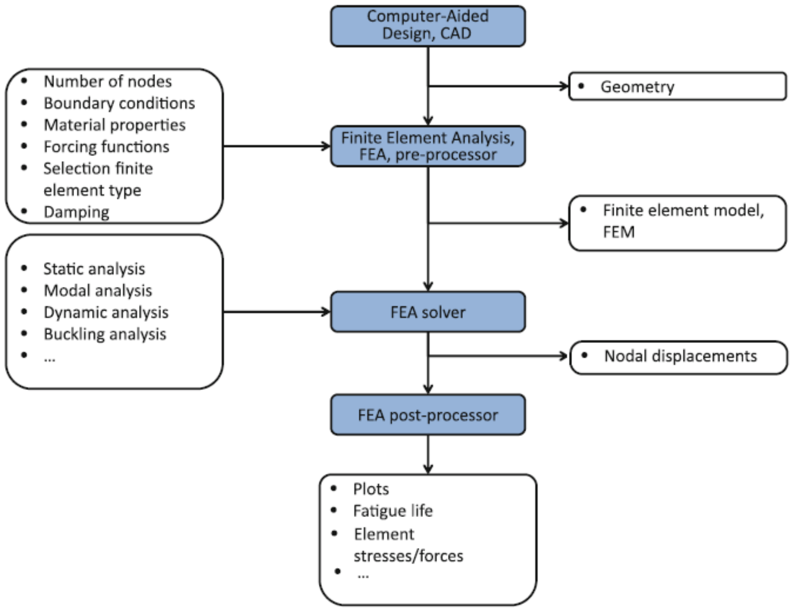

Figure 3.12: Finite Elements Method analysis flowchart (figure from Ref. [17]).

The Static Equilibrium is obtained with:

$$[K]\,\{u\} = \{F\} \qquad (3.1)$$

where:
$[K] \Rightarrow$ Stiffness Matrix is affected by the mathematical model and the governing equations chosen.
$\{u\} \Rightarrow$ Grid Points Displacements are the degrees of freedom.
$\{F\} \Rightarrow$ Applied Load Vector is affected by the boundary conditions and the loads applied to the model.

The equation of Motion is:

$$[M]\,\ddot{u}\,(t) + [B]\,\dot{u}\,(t) + [K]\,u\,(t) = \{F\,(t)\} \qquad (3.2)$$

where:
$[M]\,\ddot{u}\,(t) \Rightarrow$ Inertial Force.
$[B]\,\dot{u}\,(t) \Rightarrow$ Viscous Damping.
$[K]\,u\,(t) \Rightarrow$ Elastic Forces.
$\{F\,(t)\} \Rightarrow$ Applied Forces.

For Free Vibration Analysis without damping, such as the modal analysis, the numerical model is based on the Equation 3.2, with the term of viscous damping and the applied forces set to zero, and it is presented in Equation 3.3. The analytical solution for displacements is a function of time and it can be achieved with a cyclic response with the form Equation 3.4, an eigenvalue problem which gives the natural frequency of the structure in analyse.

The equation of Motion for Undamped Free Vibration used in Modal structural analyses is:

$$[M]\,\ddot{u}\,(t) + [K]\,u\,(t) = 0 \quad (3.3)$$

with the cyclic structure response of the structure in the form:

$$u\,(t) = A\,\sin(\omega_n)\,t + B\,\cos(\omega_n)\,t \quad (3.4)$$

where the circular natural frequency is:

$$\omega_n^2 = \frac{[K]}{[M]} \quad (3.5)$$

and the natural frequency is:

$$f_n = \frac{\omega_n}{2\pi} \quad (3.6)$$

particularly in Equation 3.4 and Equation 3.5 whit the known conditions of:
$u(t = 0)$ and $\dot{u}(t = 0) \Rightarrow$ Initial condition in order to obtain A and B.

To point out that, static analysis is a linear analysis, where is considered the infinitesimal small displacements and the linear elastic performance of the materials used, but at the dynamic this cannot be evaluated in the same form, the non-linear elastic performance of the materials must be applied [100, 120].

The process of verification and validation is a systematic process of checking the obtained results and it is performed in the post-processing phase. In this phase some questions are asked, for example if the numerical model is the right one, if it was solved correctly for the situation, if the problem has the expected results in comparison with the simplified hand calculations performed in the pre-processing phase [117].

3.2.3.2 FEA verification

Before the validation of the FEA model be accomplished by the correlation between tests and analysis, it is useful to first run a verification of the computational model in use. Typically, there are two ways to perform computational model verifications, do it through the used code or the obtained solutions [104].

The various verifications methods that are performed to the code are presented in Table 3.5, and these will be the main focus of FEA verifications in this work. The solution verifications are intended to compare the numerical and the analytical solution of the problem, but due to the difficulty of the analytical solution to be performed, in complex structures like a CubeSat; due to the obstacles in the interconnections between satellites parts, which conduct to a longer time to calculate loads effects and to the necessity of more simplifications which is associated with lower accuracy in the results obtained; and due to the necessity of analysing all the conditions from the beginning when is necessary or required, to reconsider other design this option was not approached [123].

From the Structural finite element model standard, created by ECSS are referred some verification methods like the model geometry checks or the element topology verifications that should be performed, but more methods of verification were added in Table 3.5 using information from Refs. [84, 101, 111, 124, 125].

Table 3.5: Verification methods for FEA (information in accordance with Refs. [84, 101, 111, 124, 125]).

Model Check	Criteria
Model Geometry	Unconnected nodes Coincident elements Free edges shall be expected model boundaries
Element Topology	Type of element Aspect ratio Warpage Jacobian Interfaces definition Numbering rules Coordinate system Mesh density and refinement
Rigid body motion	Mass matrix Strain energy Residual forces check
Static Analysis	Residual load vector work
Stress free Thermo-elastic deformation	Maximum acceptance values of isothermal stress and rotation
Modal Analysis	Free-free check

In the first two points, Model Geometry and Element Topology of the Table 3.5 are the normal considerations to have in attention when a FEM model geometry is created, and ordinarily the software used to create the model provide tools to check these parameters [126]. On the other hand the type of element, the conditions when is appropriate to use a spring, a bar or a beam, the way to represent a bolt or a riveted, the different interconnections and interfaces between elements should be confirmed by the structural analysis engineer, and check if the model represents the problem in a viable way.

Other points to have in attention when an analysis is being prepared are the numbering rules (each subsystem, unit, part, ... should have a specific number identification), the coordinate system where the model was built (being important to create the model in agreement with the reference frame, to be always the same one to be considered [78]), and the mesh refinement.

The mesh density and refinement, are related with the solution convergence and change in agreement with the type of analysis that is performed. Also, the element type, the elements quality, the connections between elements, the material behaviour, among others parameters change the evaluation of the mesh density and refinement [127]. In an effort to obtain a reasonable solution, trade-offs evaluating the modelling time, the analysis accuracy (associated error) and the computation time must be made.

An example of a study performed in order to obtain a suitable mesh density, and to achieve a solution convergence is described in Ref. [127]. It calculates the error in the maximum von Mises stress in function of the element number along the plate length, in a simple case of a static analysis of a rectangular steel plate (300 mm x 200 mm and 3 mm of thickness), fully constrained at one end and at the other end a 1 Nm moment is applied. As recorded in Figure 3.13 the solution represents an optimal combination of accuracy and efficiency with a number of 10 elements in the longest side.

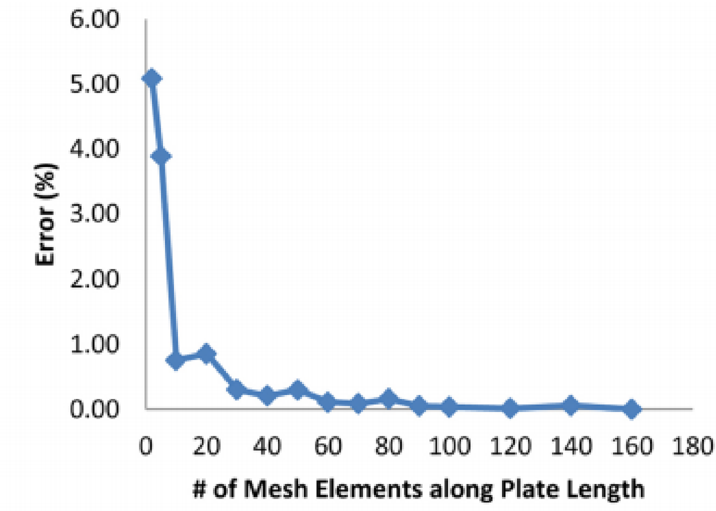

Figure 3.13: Error in the maximum von Mises stress in function of the number of element along the plate length (figure from Ref. [127]).

In agreement with the rigid body motion, some are the verifications that can be performed to the model. The rigid body motion is defined by the matrix Φ_R, of displacements based on the model geometry rather than stiffness or eigenvector calculations (a more precise definition can be found in Refs. [128, 129]). The six DOF's are defined with respect to a selected reference point, but rotations around another point, different from the centre of gravity of the model could be considered if necessary (an example, where this change of reference plane is applied, is the analysis of a couple load analysis, with the reference in the launcher vehicle instead of the satellite gravity centre).

The goal of the rigid body motion mass matrix check is to calculate the M_r, following the Equation 3.7 and evaluating if the resultant values match with the ones of the initial problem [84].

The Rigid body motion mass matrix is:

$$M_r = \Phi_R^T \, M \, \Phi_R \tag{3.7}$$

$$M_r = \begin{bmatrix} m & 0 & 0 & 0 & -m\,z_{cog} & m\,y_{cog} \\ 0 & m & 0 & m\,z_{cog} & 0 & -m\,x_{cog} \\ 0 & 0 & m & -m\,y_{cog} & m\,x_{cog} & 0 \\ 0 & m\,z_{cog} & -m\,y_{cog} & I_{xx} & I_{xy} & I_{xz} \\ -m\,z_{cog} & 0 & m\,x_{cog} & I_{yx} & I_{yy} & I_{yz} \\ m\,y_{cog} & -m\,x_{cog} & 0 & I_{zx} & I_{zy} & I_{zz} \end{bmatrix}$$

where:
$M_r \Rightarrow$ Rigid body motion mass matrix;
$\Phi_R \Rightarrow$ Rigid body motion of the Finite Element model;
$M \Rightarrow$ Mass matrix;
$m \Rightarrow$ mass;
$I_{i,j} \Rightarrow$ Moments of Inertia, being $i, j = x, y, z$;
x_{cog}, y_{cog} and $z_{cog} \Rightarrow$ Defining the expected coordinates of the centre of gravity.

The Strain energy check is another check using the rigid body motion matrix. It is performed calculating the values of the matrix E_r, obtained with Equation 3.8, that theoretically are zero, it is accept if its do not exceed $10^{-3}\ J$ [101]. Connected to the strain energy check is the residual forces check, that is performed following Equation 3.9, being the residual nodal forces theoretically values equal to zero, but it is allowed to go until 0.1 N and 1.5 x N x m [111]. Both of these verifications are in agreement with the premise that, should be computed and reported for all the nodes and for each of the six rigid body motions.

The Rigid body motion strain energy and the Rigid body motion residual nodal forces are formulated in agreement with Ref. [123]:

$$E_r = \frac{1}{2}\,\Phi_R^T\,K\,\Phi_R \tag{3.8}$$

$$F_r = K\,\Phi_r \tag{3.9}$$

where:
$E_r \Rightarrow$ Rigid body motion strain energy;
$\Phi_R \Rightarrow$ Rigid body motion of the Finite Element model;
$F_r \Rightarrow$ Rigid body motion residual nodal forces;
$K \Rightarrow$ Stiffness matrix.

In Static analysis verification is presented the unit load check. This check is performed in a generic case where is applied a static unit load to the model and compared the external load resultant reaction force at the constrained nodes. In the static analysis also the residual load's vector check should be performed as enunciated at Equation 3.10. The allowable ratio, ε in residual loads vector check cannot overcome the 10^{-8}, once theoretically this value should be zero [84, 126].

The Residual loads vector work is obtained with:

$$\begin{gathered} \delta F = K\,u - F \\ \delta W = u^T\,\delta F \\ W = \frac{1}{2}\,u^T\,F = \frac{1}{2}\,u^T\,K\,u \\ \varepsilon = \frac{\delta W}{W} \end{gathered} \tag{3.10}$$

where:
$\delta F \Rightarrow$ Residual force vector;
$K \Rightarrow$ Stiffness matrix;
$u \Rightarrow$ Displacement vector;
$F \Rightarrow$ Load vector;
$\delta W \Rightarrow$ Residual work;
$\varepsilon \Rightarrow$ Work ratio.

In the stress free thermo-elastic deformation check is evaluated the stress and rotations values when the model is built with a homogenous and isotropic material (normally is used aluminium alloy with all the thermal coefficients of expansion as well as the Young's and Poisson's modulus changed to a single value) and it is assumed an isothermal expansion (uniform temperature increase, normally is applied a $\delta T = 100K$). If the model was consistently built, should be no rotations, forces or stresses (for the case of using an aluminium alloy the maximum value of Von Mises stress allowed is $0.01MPa$ and a value of maximum rotation less than $10^{-7}rad$ [101,123,125].

When a modal analysis is constructed, is important to have a particular attention to the conditions of it since, a significant part of the definition of the dimensioning of the structural components depends on it. The check to have into account in this model is the free-free check. This check uses the rigid body modes of the model, and claim that the first six free model frequencies have to be less than 0.005 Hz [84].

3.2.4 Structural Factors of Safety for Spaceflight hardware

The structural factors of safety are used for design, dimensioning, and verification of spacecraft hardware and it is used with the aim of covering uncertainties in materials mechanical properties; to prevent errors in the final product manufacture and assembly; to avoid the possibility of a non predicted event exceed the maximum expected load level and damage in an invariable way the structure; among other reasons.

Because of it the factors of safety and the margin of safety (calculated using Equation 3.11, for each part of a satellite are one of the most important conditions to bear in mind when is desired to optimize a space structure. If an adequate configuration, design, material selection, element stiffness is performed a positive MoS will be obtained but if a negative value is achieved then the project requirements were used poorly.

The Margins of safety are calculated with:

$$MoS = \frac{design\, allowable\, load}{design\, limit\, load\, \times\, FOS} - 1 \tag{3.11}$$

considering:
$MoS \Rightarrow$ Margins of Safety;
$FOS \Rightarrow$ Factor of Safety.

Such as presented in Figure 3.14 and outlined in Figure 3.15, some are the factors that must be followed in the design and configuration of a spacecraft. The safety factors can change in agreement with the levels of the project, with the type of verification that will be executed, the family of the material in use, among others parameters.

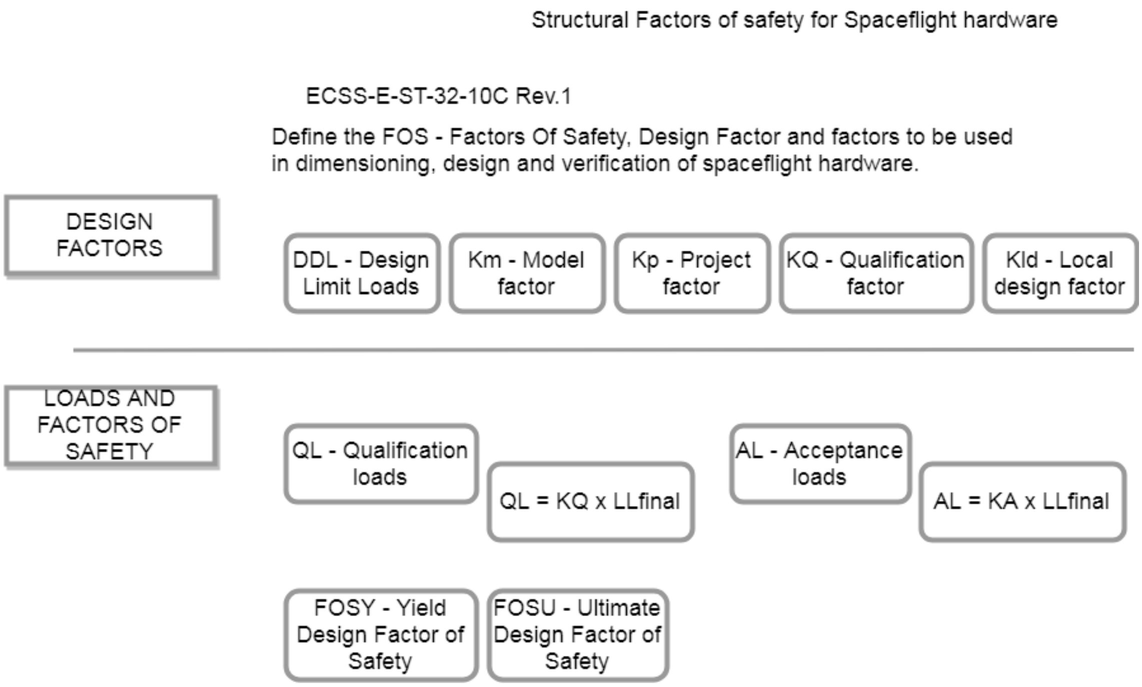

Figure 3.14: Structural factors of safety (figure compiled in agreement with Ref. [93]).

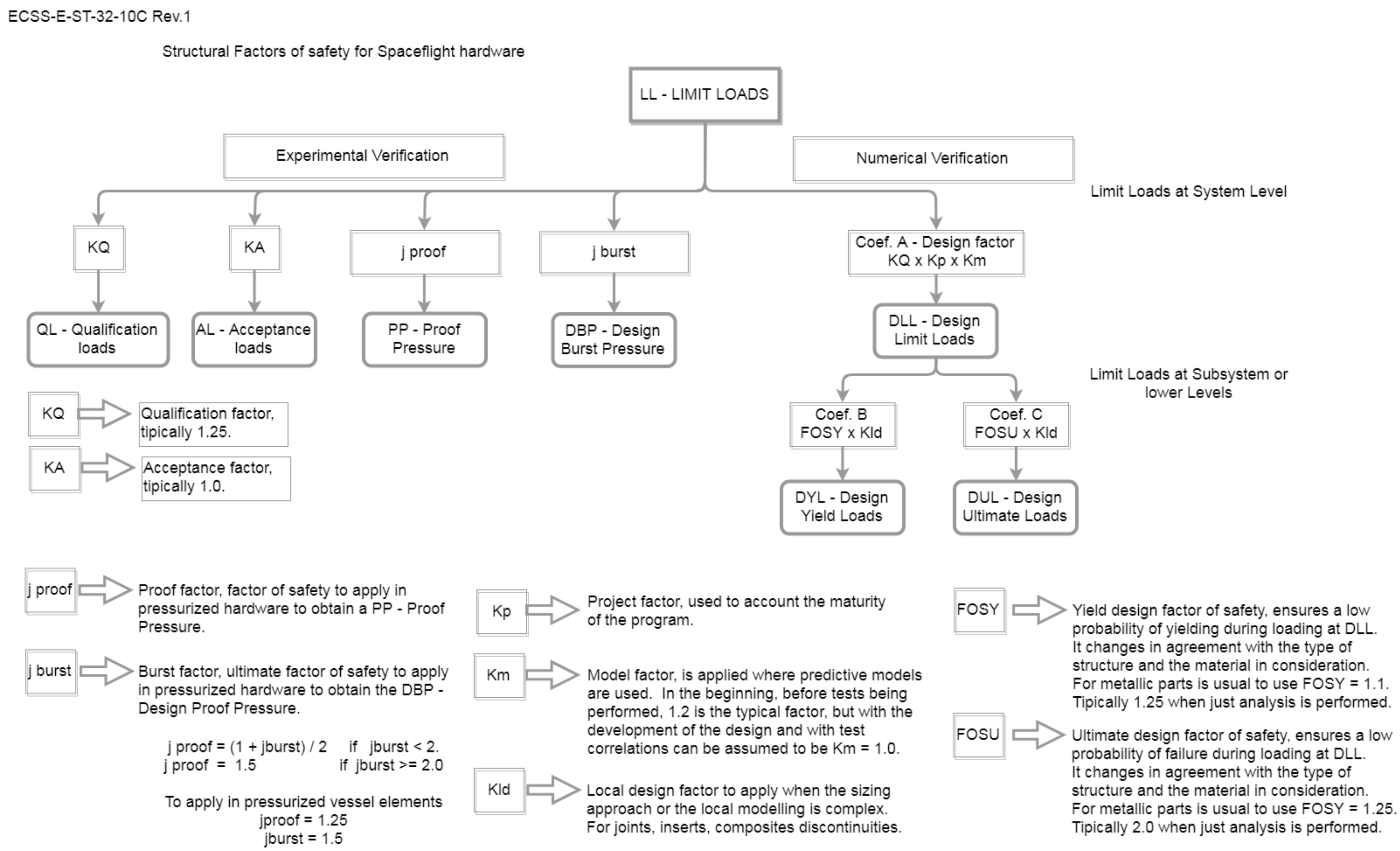

Figure 3.15: Structural factors of safety for satellites, logic application (figure in concordance with Refs. [52, 83, 93]).

In Table 3.6 part of the FOS to apply in MECSE are described. It was selected looking to the standards from ESA, more precisely Ref. [93] and NASA requirements expressed at Refs. [52, 130].

The FOS were selected to be applied in metallic parts and no other material. To note that, the materials for space use also need to be in agreement with some requirements (an overview of the main material requirements is presented in Figure D.5, in agreement with the standard Materials, Ref. [88]). If it is intended to be used other type of material, the factors of safety and other conditions should be reviewed. In the same way, the factors of safety in design and test for application in threaded fasteners have to be studied and compiled, in agreement with Ref. [92] and Ref. [130] or the conditions apply the FOS when is used a bolt or a riveted, can be consulted in Ref. [89].

Table 3.6: Factors of safety for satellite application, in metallic parts (information in agreement with Refs. [52, 93, 111, 130]).

Load type	**Requirements**			
	KQ Qualification factor	KA Acceptance factor	FOSY Yield design factor of safety	FOSU Ultimate design factor of safety
Ground, Handling and Transportation loads	2	-	1.1	1.25
Global flight loads	1.25	1	1.25	1.4
Random and Accoustic	N/A	N/A	1.6	1.8
Internal pressure	1.5	1.25	1.1	1.25
Thermal loads	N/A	N/A	1.1	1.25

NOTES
Internal pressure with reference to pressure vessels.
At Internal pressure load the KA is assumed to be the jproof - Proof factor, and the KQ is assumed to be the jburst - Burst factor.
N/A - Not Applicable, some of the KA and KQ are not specified once it was specified in test levels and duration.

3.2.5 Analysis and Test correlation

To complete chapter 3, it is necessary to define the correlation criteria between analyses and tests, and understand how can the FEM model be validated with the test results. With the intention on the final, confirm the main purpose of the experimental verification, which is obtain part of the satellite validation. Table 3.7 refers the interconnection between analyses and tests, but the definition of the acceptable deviations from both results will be explained below.

The first correlation to be checked is the correlation between the static experimental model and the numerical model. Once, it is the first test to be enunciated in the test sequence presented in subsubsection 3.1.3.2. The validation of the numerical model is granted if the model represents in a viable way the satellite structure, it is achieved mainly if the load path between the analysis and the experimental verification match, with it is intended to mention that the evolution of the points localization where is detected the higher and lower stress and strain distribution corresponds between the numerical model and the experimental verification [123].

The others test previously indicated to the acceleration tests in subsubsection 3.1.3.2, are not being considered here since, the physical properties is an inspection, performed at the beginning of the experimental model tests, and it is also conducted in the end of the experimental verification, with the goal of evaluating if the test model has the characteristics, in agreement with what was defined in the mathematical model and in the TSPE (by characteristics is meant, the mass, structure stiffness, internal and external connections, dimensions, among others parameters, the allowable deviations of these parameters are present in Table 3.3 - Mass properties).

Table 3.7: Analysis and test correlation (information in accordance with Ref. [87]).

Load	Verification by analysis	Verification by test
Static and Quasi-static	Static Analysis	Static Test Sine burst Test Sine vibration Test
Low Frequency Vibrations	Frequency response Analysis Transient Analysis	Sine Vibration Test
Broadband Vibrations	Random vibration Analysis Vibro-acoustic Ananlysis	Random vibration Test Acoustic noise Test
High Frequency Vibrations	Transient Analysis Shock propagation assessment	Shock Test
Pressure	Static Analysis Thermo-functional Analysis	Proof pressure Test Thermo-functional Test
Thermo-elastic	Thermo-elastic Analysis	Thermoa-elastic Test

The full functional and performance test is conducted to the software and equipment verification, and to the correct functioning of them. It is also mandatory the realization of these tests at the end of the mechanical experimentation. The humidity test is another inspection, performed to assure that the test facilities conditions are in agreement with the TSPE, in this case if the relative humidity of the air is at the expected range level [60].

The leak is an inspection, used to confirm if the structure does not have any slit. Linked with the leak inspection are the proof pressure test, the pressure cycling test and the design burst test. These last three tests are performed in critical items of the structure, and not in the entire model.

To note that the correlation criteria just can be valid if the model representation has the same conditions in comparison with the experimental tests and with that is intended to refer the CubeSat position, the boundary conditions, the interfaces connections, the mechanical environment and the materials used. The tests just can be performed and validated if all of these topics are in agreement with the test documentation indicated in subsection 3.1.3. With emphasis to the TSPE - Test Specifications, the TPRO - Test Procedures, the tests facilities and tests calibration in agreement with the quality and safety management system, the cleanliness and contamination (defined at the ECSS-Q documents, product assurance standards), and the test accuracy presented in Table 3.4.

The sine or sinusoidal vibration tests are used, mainly to verify the calibration of the instruments, if it is working with the expected responses or not [60]. The identification of the natural frequencies and the mode shapes are the significant verification, to perform in the dynamic model analyses, being the broadband vibrations tests and the shock test the most demanding verifications and because of that, it is the principal attention of the dynamic validation.

Important to present that some modes can be coupled which leads to a higher response than was expected. This happens mainly in the random vibration and in the acoustic tests where the frequency intensity is higher, leading to larger variations in the tests results in comparison with the analyses results. This is the reason why most of the time the notching technique is used, allowing the study evaluation of the modes and verify if the response is evolving in a predicted way and do not subject the structure to overtesting, with the coupled of modes [111, 113].

The modal survey standard defines a tight deviation on the dynamic structures responses (natural frequencies and mode shapes) being acceptable a deviation of only 0.5% in the natural frequencies and 5% in the mode shapes, in order to assure the validation of the model. These values can vary slightly in agreement with the modal method used to identify the required solutions, being the same ones presented in Ref. [94]. To note that, more than one of the methods are mandatory to be used like can be consulted in Figure 3.17 - TEST, TEST-ANALYSIS CORRELATION.

In terms of thermal model validation, the analyses and the tests correlation to have in mind are similar to the ones at the static model validation. It is important to have in count the deviations from the maximum and minimum thermal stress and examine the thermal load path.

All the conditions presented earlier, to be considered for validation of the numerical analyses were compiled in concordance with Refs. [52, 74].

Apart from it, others requirements are defined in Ref. [94] and expressed in Figure 3.16 and Figure 3.17. These figures define different requirements for dynamic test and the way that these should be performed, in order to be possible the correlation between the experimental verification and the numerical verification in the dynamic analyses.

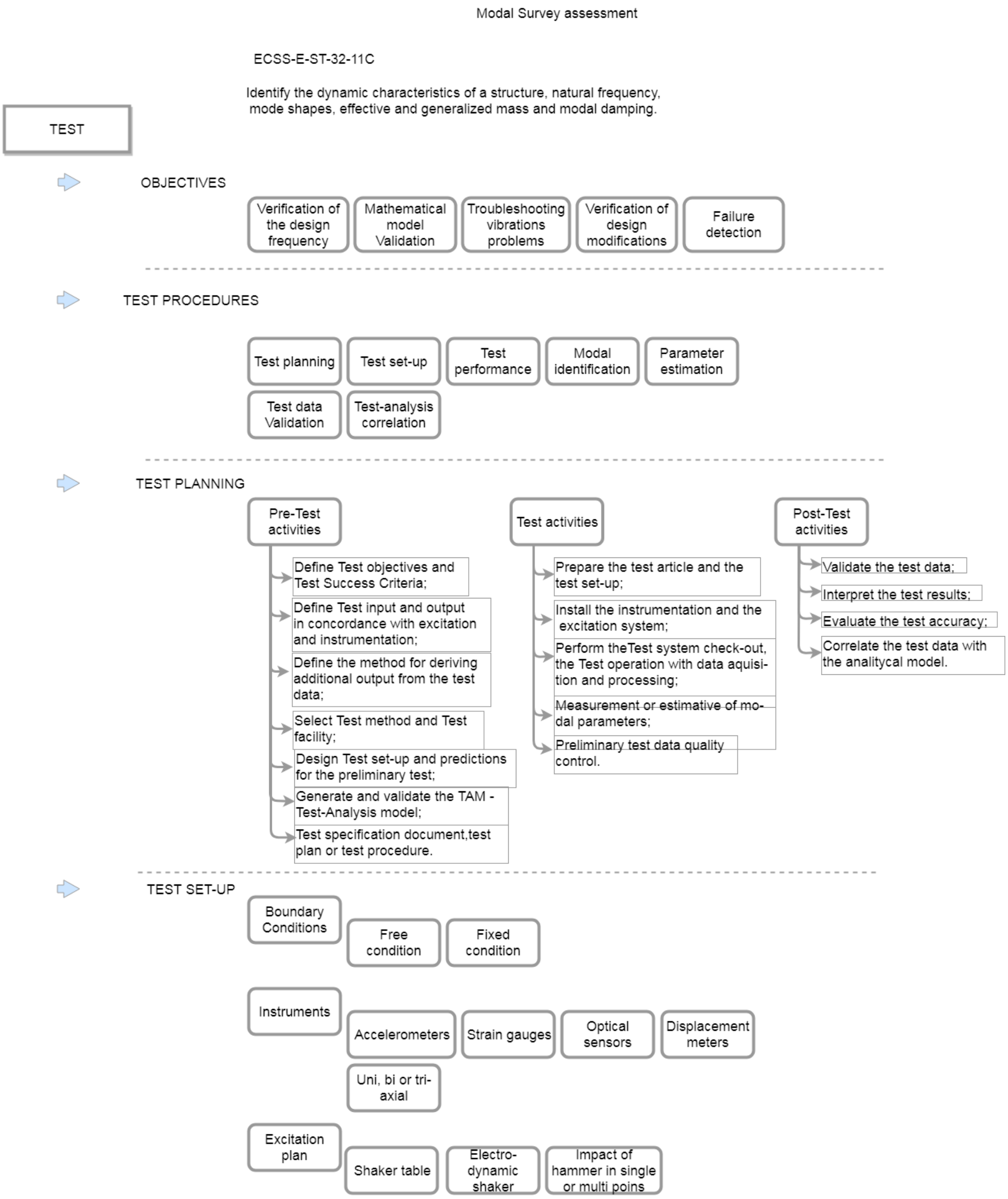

Figure 3.16: Modal survey assessment (figure compiled in concordance with Ref. [94]).

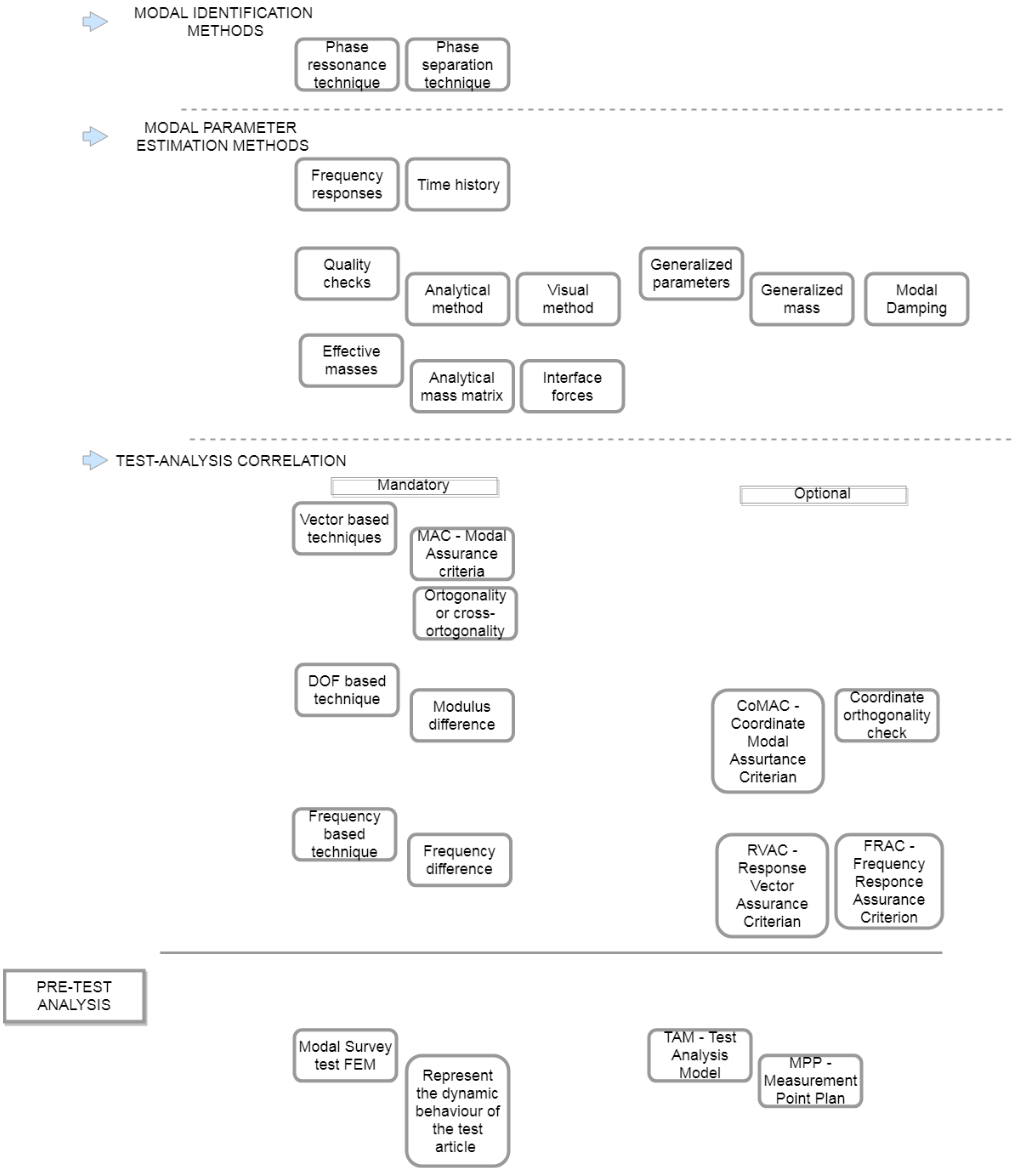

Figure 3.17: Modal survey assessment, continuation (figure compiled in concordance with Ref. [94]).

Chapter 4

MECSE Case Study

In section 2.3 - Design requirements, the different types of loads to be implemented in a CubeSat structure verification were identified; in section 3.1 an overview of the validation process was presented in a management point of view; in section 3.2 it was mentioned the verification process of a satellite structure. It is now necessary to propose a possible verification process for MECSE specify the magnitude of the limit loads that should be used in the analyses and tests.

In this chapter 4, the structural limit loads will be identified in agreement with the launch vehicle requirements, and the way the satellite will be transported to the launch facility. Reasons that lead the process of selecting a launch vehicle for a subject of extreme importance, in the configuration and structural characterization of any satellite.

The preliminary structure of MECSE has been designed for use in Vega launch vehicle, already indicated in the previous chapters however, an overview of other possibilities is presented in this chapter. Whit the ambition of introducing a more suitable launch vehicle to MECSE project, as well as with the strong desired to formulate a summary among some of the launch vehicles available on the market at this moment. This will result in the proposal of the most demanding conditions that a satellite must overcome if the intention is to launch with one of the selected vehicles.

The launch vehicle industry is another sector of the space market, which has been increasing significantly its number of suppliers, with around 100 small launch vehicles in development. These statistics represent the reaction of the increase in number of developing payloads, especially the micro and nanosatellites that are looking for competitive prices for launch [131].

Even though the intention is not to launch with any of the launch vehicles presented in this dissertation, since there are plenty of options, the conditions for a 3U CubeSat should not change drastically and with this, the use of the WCS that will be presented in section 4.3 is a possibility, also due to safety factors and margins of safety applicability during the design and numerical verifications that should convey confidence in the MECSE project in progress.

The space vehicles were selected to compile examples of launch vehicles all around the world. It has been selected launchers with extensive experience in launching successful satellites, such as the Russians, Rockot and Dnerp-1 [132, 133], which have intrinsically a well defined mechanical environment. It was selected launch vehicles that are under development, with proposals that change the conventional launch profile such as the LauncherOne, from Virgin Orbit USA, or the Spanish Bloostar, from Zero 2 Infinity [134, 135]. It was selected launchers from specialized organizations in launch CubeSats and small satellites, like for instance the launcher Electron from the American Rocket-Lab or the Vector company, whit the launcher Vector-R [136, 137] and it was also analysed a reusable microlauncher, the Arion-1, a launch vehicle under development from the Spanish PLD Space [138].

Another option available to MECSE is to be launched from the ISS - International Space Station. Due to MECSE being a CubeSat its defined orbit being 350 km at apogee and 52.6º of inclination the launch from ISS is possible [139]. If using the ISS is a possibility, then the Soyuz launch vehicle and the Falcon 9 are also a possibility, since its success in ISS coupling missions are always known [140, 141]. For this reason, these two options were studied alongside with the former launch vehicle Vega.

An additional solution besides launching a CubeSat from a conventional launcher, and which it is identical from launching a small satellite from the ISS is to launch the MECSE from an external platform coupled to the ISS [142, 143]. From a mechanical and structural point of view, the difference between the ISS option or an external ISS platform is minimum, the difference would be detected in the services that the company owner of this platform could provide in this case for MECSE project [144]. An under development external payload is Bartolomeo's platform, it will begin to accommodate payloads at the most at the beginning of 2020, a possible opportunity for MECSE [145].

Even before the launch event it is necessary to build and transport the satellite to the launch facility and for that the position where the launch facility is located is important. This subject will be reported in subsection 4.2.1, following a crucial and concise description of preferred launch vehicles suppliers. The launch mechanical environment will be defined in subsection 4.2.2 and still in this chapter 4, in section 4.3 a reference will be made to the most demanding loads of the selected launch vehicles.

The values that will be presented in this chapter 4 are always in reference to the WCS, and this should take into account the perfect coupling between CubeSat and P-POD, or another deployer, it is assumed that any load will be amplified or damped by the deployer, moreover the load will be applied directly to the structure, which is the definition of the CLA - Coupled Load Analysis.

4.1 Launch Vehicles

In this section will be expressed the different launch vehicles and some of the main requirement, with concerns in the functional requirements, with the identification of the minimum natural frequency; and interface requirements, describing how the external interface between the launch vehicle and CubeSat deployer interaction could be made. References to the final orbit and the launch localization will also be presented below.

The minimum allowed frequencies for a satellite structure are not well defined by the User's Guides from the following launchers: Arion 1, Bloostar, Electron, Falcon 9, LauncherOne and Vector-R. In response to that in section 4.3 - Minimum allowable frequency an identification of the WCS for the minimum resonance will be provided and it can be used as a reference for these launchers that do not define a minimum natural frequency to the satellite structure.

4.1.1 Arion 1

Arion 1 is one of the Spanish microlaunchers being developed by PLD Space, with a single stage launch profile [138]. The purpose of this launcher is to offer a suborbital launching service for small satellites and payloads, being developed to deploy payloads up to 170 km at apogee, which is not the desired orbit for MECSE, but a possible evolution of this model, for a higher range launcher (which is a goal for PLD Space) can be used in the particular case of MECESE [138]. Therefore, if the main objective of the MECSE project, beyond the educational objective, changed to the technology's mission of proving that the MHD/EHD device for plasma layer manipulation could overcome the RF blackout on reentry of a spacecraft into the Earth's atmosphere, Arion 1 could be used. Arion 2 is another product from PLD Space that could be studied, but due to the preliminary phase of development of this product, the information is very limited. The objective of this second PLD Space launcher is to launch a 150 kg payload for LEO, more precisely for 400 km of altitude, which would be suitable for MECSE [131].

PLD Space's headquarters are based in Elche, where pre-flight acceptance tests are conducted, the launch site is at El Areonosillot [138]. The interface between CubeSat, MECSE in this case and the launch vehicle need to be discussed, since this launcher is optimized to transport a single payload into orbit instead of providing a "rideshare"[1].

4.1.2 Bloostar

Another launch vehicle analysed was the Spanish Bloostar. It is a launcher under development, dedicated to nano and microsatellites up to 75 kg, claiming advantages for the customer, such as the lesser need for expensive and time consuming qualification tests [135]. Zero 2 Infinity designed this launcher to place a satellite or a satellite constellation into LEO, offering various solutions in the mechanical interaction between payload and launch vehicle [135]. The mechanical environment is not well defined due to the preliminary evolution of the project, however the ascending profile of this launcher is characterized by three stages and based on the Rockoon method, changing the first stage of a conventional rocket into a helium balloon, which reduces vibrations and shock loads that are required to be overcome and to put a satellite or satellites into orbit. This method allows a flexibility at the launch location [135, 147].

4.1.3 Dnerp-1

Dnepr-1 is a product from Kosmotras, a Russian company that provides launch solutions since 1999. This launcher is used for injections of a single satellite into orbit or sets of satellites into LEO orbit, more precisely between 300 and 900 km at apogee with inclinations of 50.5; 64.5; 87.3 or 98 degrees that MECSE can be adapted for [133]. It is characterized by a three stage rocket modified from the liquid-fuelled SS-18 ICBM, in addition to the spatial head module in line configuration [133, 148].

This launcher is not used regularly to launch lightweight spacecraft like 3U or 6U CubeSats, as it

[1] "Rideshare" is a term used when a launcher supplier chooses to split the launch payload area into multiple compartments and charge a specific price for each area instead of selling the complete payload [136, 146]

does not have adapters optimized for this type of satellites, which could be an obstacle for MECSE. The minimum natural frequencies allowable for the satellite structure are 20 Hz on the longitudinal axes and 10 Hz in the lateral [133].

4.1.4 Electron

Electron is another launch vehicle built to launch constellations of small satellites. It is capable of launching 150 kg into a sun-syncronized orbit 500 km at apogee, but also capable of launching payloads from 300 to 700 km at 45º of inclination [136]. The RocketLab Complex in New Zealand or the USA facility are the launch points for this two stage launch vehicle [149]. It is prepared to easily couple CubeSats and its standards deployers.

4.1.5 Falcon 9

This launcher built by SpaceX is one of the launch vehicles with successful flights to ISS, which has become a suitable option to be applied in the MECSE project. Falcon 9 is a two stage launcher capable of delivering satellites to LEO, 200 to 360 km at apogee [141]. Space X's launch facilities are located in Florida, being this launch facility dedicated only to launch the Falcon 9, and another facility in California. The ability to accommodate CubeSats needs to be discussed with SpaceX, or with a business consulting company that splits the payload area into a "rideshare" [141].

4.1.6 LauncherOne

LauncherOne is a completely different launch vehicle from the rest of the launchers featured in this dissertation, due to its launch profile. It has a launch profile with the first phase, where the launcher is carried by a Boeing 707-400 until an altitude of approximately 10000 meters, and after it the rocket, LauncherOne is fired and it proceeds to its two propulsion stage until the final orbit, with a maximum of 1200 km altitude [134]. The great advantage of this launch vehicle is the availability to be launched almost from every position on earth, and capable of delivering payloads into a wide range of inclinations, in addition to the high compatibility with various CubeSat dispensers [134]. This option, although original does not bring many benefits in terms of mechanical loads nevertheless, this theme will be evaluated later in section 4.2.

4.1.7 Rockot

Rockot is one of the launch vehicles developed by Eurorockot in a German-Russia partnership, in the late twentieth century. The Rockot is a three stage launcher built at the base of the SS-19 missile. It is launched from Plesetsk Cosmodrome in Northern Russia, and it has LEO as the desired orbit to be achieved. It is capable of carrying about 2000 kg into 200 km of altitude [132, 150]. The applicability of Rockot on MECSE, and in CubeSats in general would be difficult due to the lack of preparation of this model to accommodate microsatellites and CubeSats [132].

The minimum allowable frequencies for the structure are 15 Hz in lateral axes and 33 Hz on the axial ones [132].

4.1.8 Soyuz

Soyuz is a European launch vehicle of Arianespace, and it is one of the primary transports for ISS [140]. This launcher can distribute payloads into a wide range of orbits and inclinations and it is widely used for satellites within a wide range of missions, from earth observation, meteorological and scientific missions [151]. It is a three stage launcher with a capacity to incorporate 4850 kg of payload when the goal is to deliver it at an altitude of 920 km and 52º of inclination [140]. It is a suitable option for MECSE, whether it is integrated into a ”rideshare”, with a multi CubeSat deployer, or embedded in a standard deployer.

The minimum allowable frequencies for the structure are 15 Hz in lateral axes and 35 Hz on the longitudinal ones [140].

4.1.9 Vector-R

Vector is a family of small launch vehicles formed by Vector-R and Vector-H. Vector-R is designed to provide fast and frequent launches to LEO [152]. This launcher launches approximately 65 kg with a two stage rocket at one of two launch sites Cape Canaveral Air Force Station, Florida for low inclinations such as MECSE, and Pacific Spaceport Complex in Alaska for high inclinations - from 60º to 102º [137]. This launcher is designed to launch small satellites and CubeSats, which would facilitate the mechanical interconnection between MECSE, deployer and launch vehicle.

4.1.10 Vega

Vega was the lastest launch vehicle analysed. It is a three stage vehicle, as can be seen in Figure 1.8, built to deploy payloads into LEO, with a maximum altitude of 1400 km [153]. The facility sited in French Guiana is the operational launch centre of Arianespace and where all launch vehicles of this company are launched [49]. The launch opportunity analysed for Vega was taken into consideration from Ref. [153], where it is recognized the well defined and optimized mechanical interface for CubeSats. This particular launch opportunity will be the reference and the baseline for the rest of this work, when Vega is specified.

The minimum allowable frequencies for the structure are 45 Hz in the lateral axes and 90 Hz in the longitudinal ones [154].

4.2 Mechanical Environment

In this section 4.2 a description of the most demanding conditions in the mechanical environment is presented, defining some of the limit loads for MECSE application.

Note that all the dynamic loads are statistical approaches and never a precise measurement of the load. An explanation about the source of each load category can be consulted in Appendix E.

4.2.1 Ground, Handling and Transportation

Normally ground, handling and transportation of satellites are determined by the type of carrier that the satellite is transported to the launch site. It depends on whether the transportation is done by water, air or on the ground by a truck or train. In addition, the GSE involved has an important role in the handling and construction of a satellite. The equipment used to assemble and transport the spacecraft needs to be taken into consideration, once it may create more loads on the structure than it will suffer for the rest of its life [155].

Sometimes random vibration can also be detected in this phase, when the satellite is transferred by truck or in a railway, but the main random vibration loads are applied at launch [138].

A summary of the ground, handling and transportation loads is presented in Table 4.2. Note that the information was collected from the different launch vehicles user's guide. If it is not clearly specified in the launcher user's guide, the values adopted are the general case presented in Table 4.1, for ground and transportation using a truck for longitudinal and vertical load factors (± 3.5 g and ± 6 g respectively), instead the lateral load factor is adopted from the transportation on water (± 2.5 g) and these values are identified with *. These values were selected in concordance with Ref. [111, 155], since these are the worst cases of loads for ground and transportation.

Table 4.1: Ground and Transport loads (data from Refs. [111, 155]).

Mode	Longitudinal Load Factors [g]	Lateral Load Factors [g]	Vertical Load Factors [g]
Water	± 0.5	± **2.5**	± 2.5
Air	± 3	± 1.5	± 3
Ground			
Truck	± **3.5**	± 2	**+ 6**
Train (rolling)	± 3	± 0.75	+ 3
Slow-moving dolly	± 1	± 0.75	+ 2

Table 4.2: Ground, Handling and Transportation loads in different launch vehicles (data from Ref. [111] and launch vehicles user's guide).

Launch Vehicle	Longitudinal Load Factors [g]	Lateral Load Factors [g]	Vertical Load Factors [g]
Arion 1	± **3.5***	± **2.5***	± **6**
Bloostar	± **3.5***	± **2.5***	± **6**
Dnerp-1	- 1.7 / + 0.3	± 0.4	± 0.5
Electron	± **3.5***	± **2.5***	± **6**
Falcon 9	± 1	± 0.75	± 2
LauncherOne	± **3.5***	± **2.5***	± **6**
Rockot	± 0.3	± 1.5	± 6
Soyuz	± 0.3	± 1.5	± 6
Vector-R	± **3.5***	± **2.5***	± **6**
Vega	± **3.5***	± **2.5***	± **6**

4.2.2 Launch Environment

The identification of the parameters presented below was compiled using the launch vehicle user's guide. If part of the mechanical environment has not been defined in the user's guide, the information is left blank, an example of this situation is the sine vibration environment of Arion 1.

Quasi-Static Loads

Table 4.3 clarifies the quasi-static loads that are supposed to affect the coupled CubeSat and deployer. In it is revealed that the maximum magnitude of quasi-static loads occur in launch vehicles that are prepared and idealized to launch CubeSats and micro and nanosatellites, like is the case of launch vehicle Vega and the LauncherOne. Also, the launch vehicle Falcon 9, which is the launcher capable to transport more payload mass into orbit, of the launchers presented in this dissertation, required considerable loading factors.

Table 4.3: Quasi-Static loads in different launch vehicles (data from launch vehicles user's guide).

Launch Vehicle	Longitudinal Load Factors [g]	Lateral Load Factors [g]	Vertical Load Factors [g]
Arion 1	± 3.5	± 2	± 6
Bloostar	+ 6.5 / - 1	± 0.5	± 6
Dnerp-1	+ 8.3 / - 1	+ 1 / - 0.8	+ 1 / - 0.8
Electron	+ 8 / - 4	± 2	± 6
Falcon 9	**+ 8.5 / - 4**	± **3**	± **6**
LauncherOne	**+ 8 / - 4**	± **5**	± **8**
Rockot	+ 8.1 / - 1.5	± 1.8	**+ 8.1** / - 1.5
Soyuz	+ 1.8 / - 5	± 1.8	± 6
Vector-R	± 3.5	± 2	± 6
Vega	**+ 10.5 / - 14.5**	± **3**	± 3

Sine Frequency

The sine frequencies are evaluated in the axial direction and in the lateral direction, always according to the reference coordinate system of the launch vehicle [78]. Figure 4.1 represents the axial vibrations, in the range of frequency between 0 Hz and 100 Hz from the selected launcher vehicles. Apart from the axial loads, the lateral loads are elucidated in Figure 4.2, with the highest point for Vega launch vehicle loads.

Random Vibration

An overview of the random frequency loads is presented in Figure 4.3. To note that these data were collected by Ref. [52], since this is the document referenced in the launch user's guide, which must be followed if there is a lack of information about any special requirements. Being the random vibrations, apart from the shock environment, one of the most demanding requirements in the launch environment, consideration of the previously highlighted reference was unavoidable.

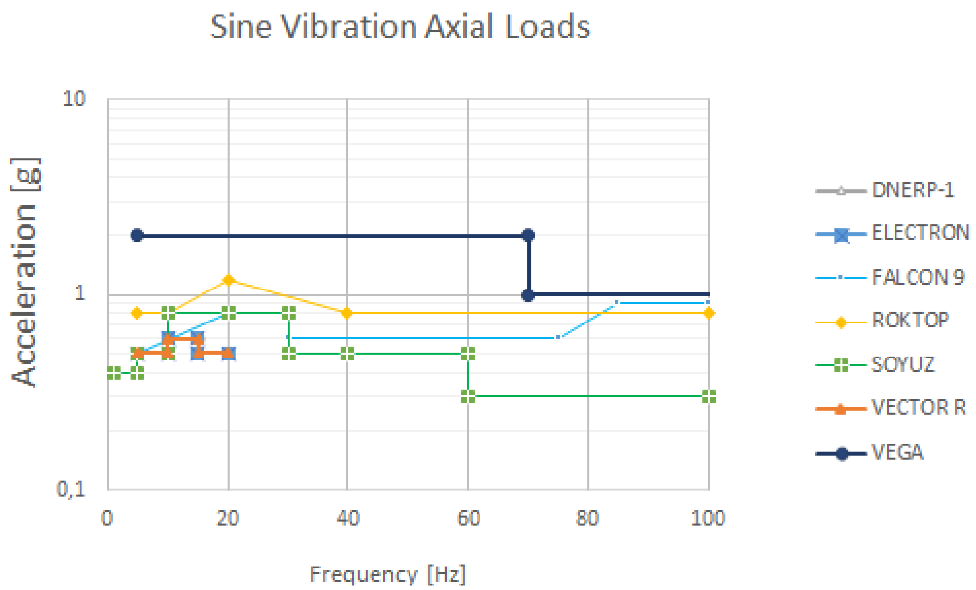

Figure 4.1: Sine Vibration Axial Loads (data from launch vehicles user's guide).

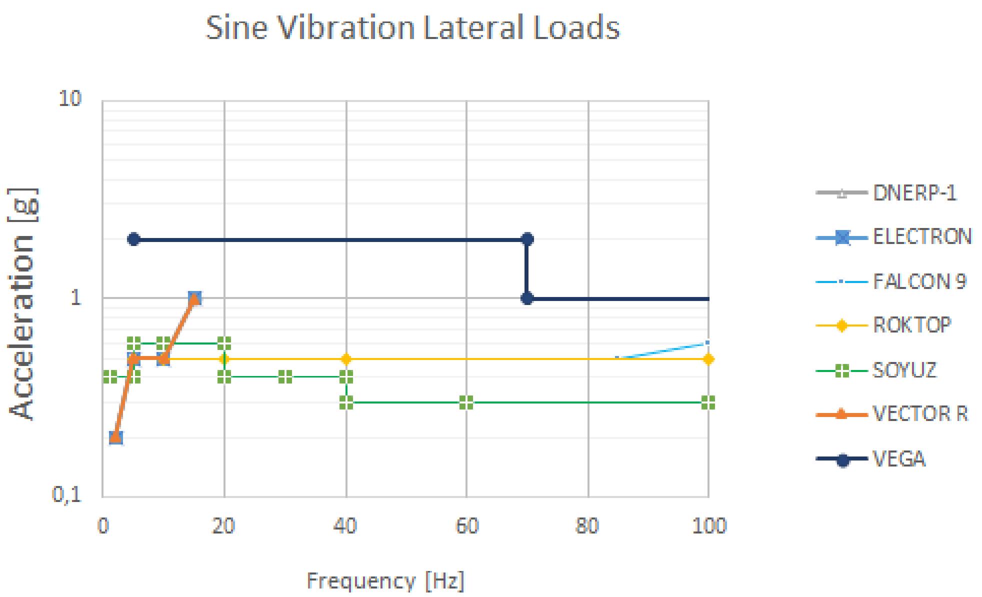

Figure 4.2: Sine Vibration Lateral Loads (data from launch vehicles user's guide).

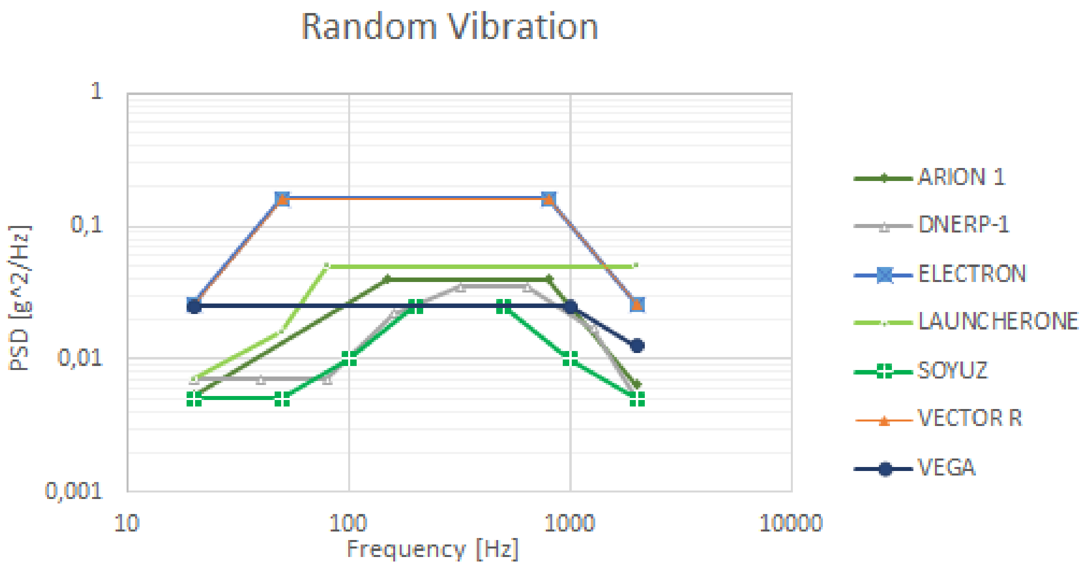

Figure 4.3: Random Vibration (data from launch vehicles user's guide).

Acoustic Noise

The acoustic loads are described by each launch vehicle are shown in Figure 4.4. Be aware of the Dnerp-1 loads, a launcher designed approximately twenty years ago and the LauncherOne suggesting that this creative idea will not be the best option, if only the mechanical environment is considered.

Some of the launchers do not define one of these two last loads (random vibration or acoustic noise), this happens since the frequency range of these loads are in the same band of frequency, they are at a broadband frequency. Moreover, the source of the load has almost the same origin, which conducts to the launch vehicles user's guides to just present one of the loads, these reasons can be noticed in section 2.3 - Mechanical Environment, and section E.1.

Shock Environment

A summary of the shock level can be found in Figure 4.5. In evidence are the Electron and the Arion 1 with the most tolerant shock loads, and once again the LauncherOne with one of the most demanding conditions, next to Dnerp-1.

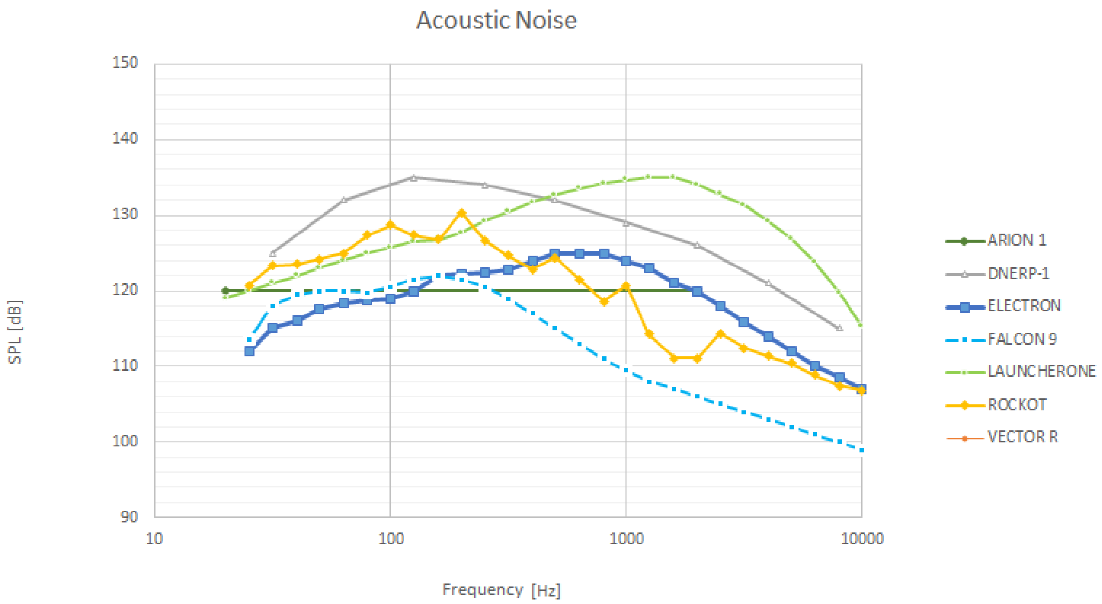

Figure 4.4: Acoustic Noise (data from launch vehicles user's guide).

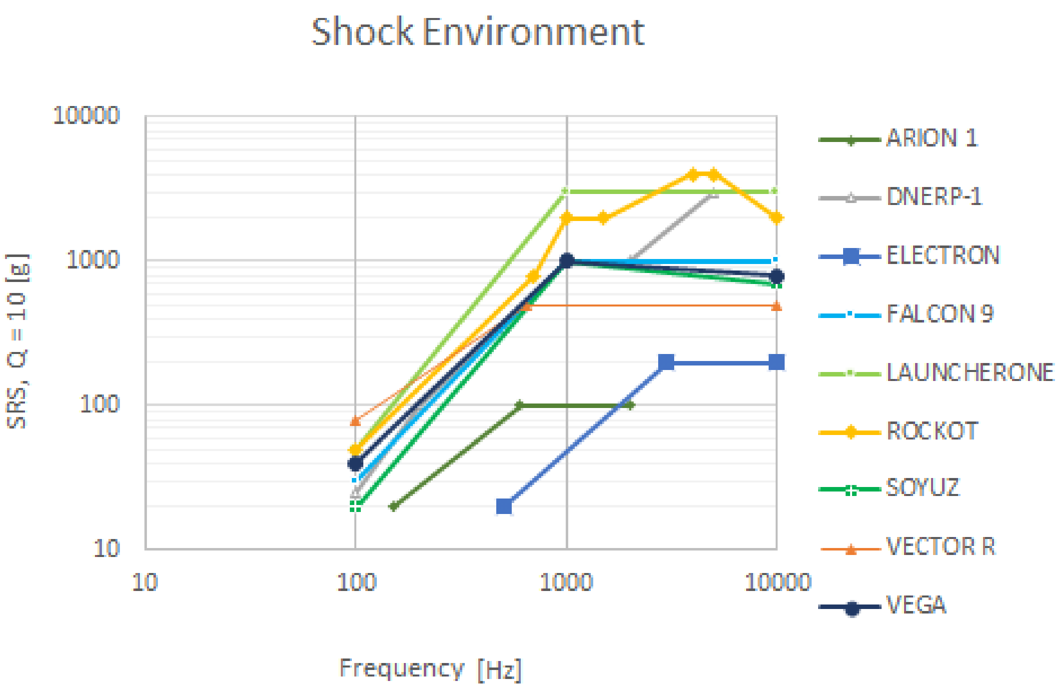

Figure 4.5: Shock Vibration Environment (data from launch vehicles user's guide).

4.3 Worst Case Scenario

This section 4.3 aims to provide a summary of the most demanding mission requirements of the launchers studied, as well as to present the FOS, that are presented in Table 3.6 to be applied in each situation in order to be applied in MECSE or in a comparable satellite.

Ground, Handling and Transportation

In Table 4.2 the cases with the highest load factors are easily highlighted, which are presented in Table 4.4. Being these the static loads that a 3U CubeSat should be prepared to overcome.

Table 4.4: Worst Case Scenario of Ground, Handling and Transportation loads (information in agreement with Table 4.2).

	Longitudinal Load Factor [g]	**Lateral Load Factor [g]**	**Vertical Load Factor [g]**
Ground, Handling and Transportation loads WCS - Worst Case Scenario	± 3.5	± 2.5	± 6

Quasi-Static loads

Analysing the Table 4.3 is outlined the load's factors from Falcon 9, LauncherOne and Vega representing the most demanding quasi-static loads. The most demanding loads are indicated in Table 4.5.

Table 4.5: Worst Case Scenario of Quasi-Static loads (information in agreement with Table 4.3).

	Longitudinal Load Factor [g]	**Lateral Load Factor [g]**	**Vertical Load Factor [g]**
Quasi-Static loads WCS - Worst Case Scenario	+ 10.5 / - 14.5	± 5	+ 8.1 / - 8

Sine Frequency

The worst case for sine frequency loads on both axes are presented in Figure 4.6 and Table 4.6. As noted earlier in Figure 4.1 and Figure 4.2 the launcher Vega is the launch vehicle, from the launchers analysed, with the most demanding conditions in terms of sine loads, and due to this, if MECSE can overcome these specifications, it will be able to support the sine environment of each launcher studied.

Table 4.6: Worst Case Scenario of Sine frequency loads values (information in agreement with Figure 4.1 and Figure 4.2).

	Frequency [Hz]	**Acceleration [g]**
Sine frequency WCS - Worst Case Scenario	5 - 70	2
	70 - 110	1

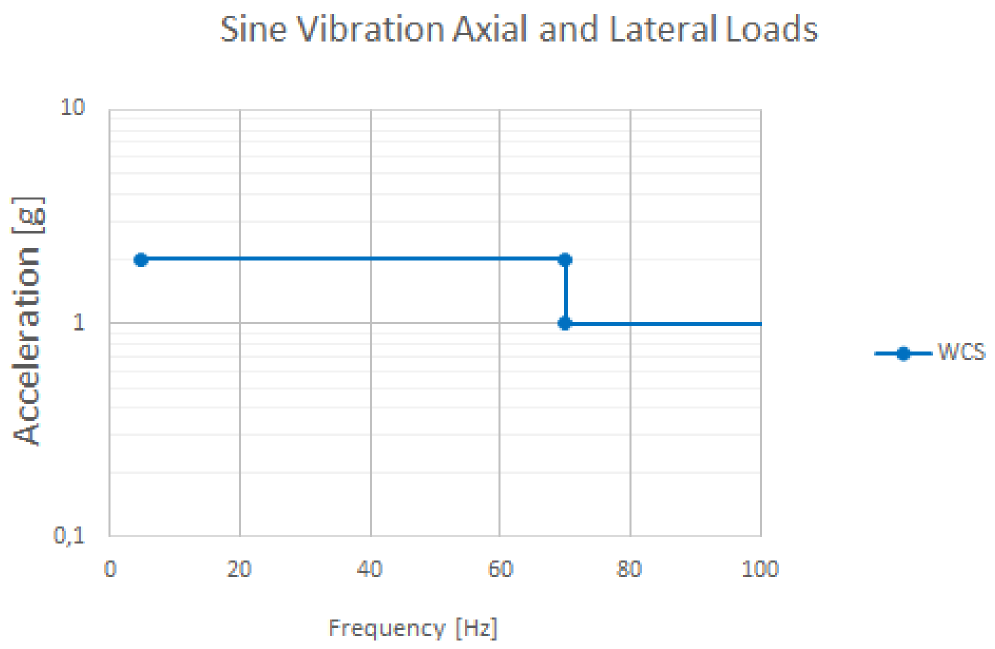

Figure 4.6: Worst Case Scenario of Sine frequency load (information in agreement with Figure 4.1 and Figure 4.2).

Random Vibration

The worst case of random frequency is indicated in Figure 4.7 and Table 4.7. It features the most demanding random vibration conditions to apply on a 3U CubeSat, with 14.1 g as the root mean square acceleration.

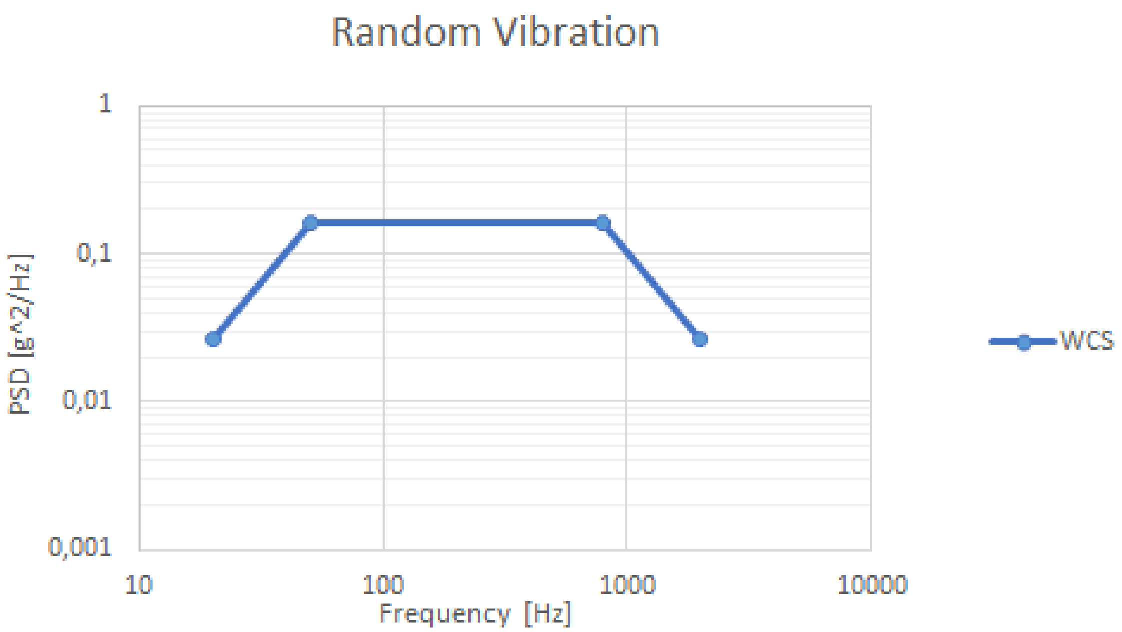

Figure 4.7: Worst Case Scenario of Random vibration (information in agreement with Figure 4.3).

Table 4.7: Worst Case Scenario of Random vibration values (information in agreement with Figure 4.3).

	Frequency [Hz]	PSD [g^2 / Hz]
Random Vibration WCS - Worst Case Scenario	20	2
	50	0.16
	800	0.16
	2000	0.026

Acoustic Noise

In Figure 4.8 is described as the worst acoustic noise scenario of the launch vehicles studied in this dissertation. The sound pressure levels are presented in Table 4.8 with an octave per frequency, with an overall sound pressure level of 141.4 dB, and a sound pressure reference of 20 uPa, the audible human sound level. A slight change in values from Figure 4.4 can be identified but it happens once it was chosen to present the WCS of acoustic noise at one octave of frequency and not at a third of an octave.

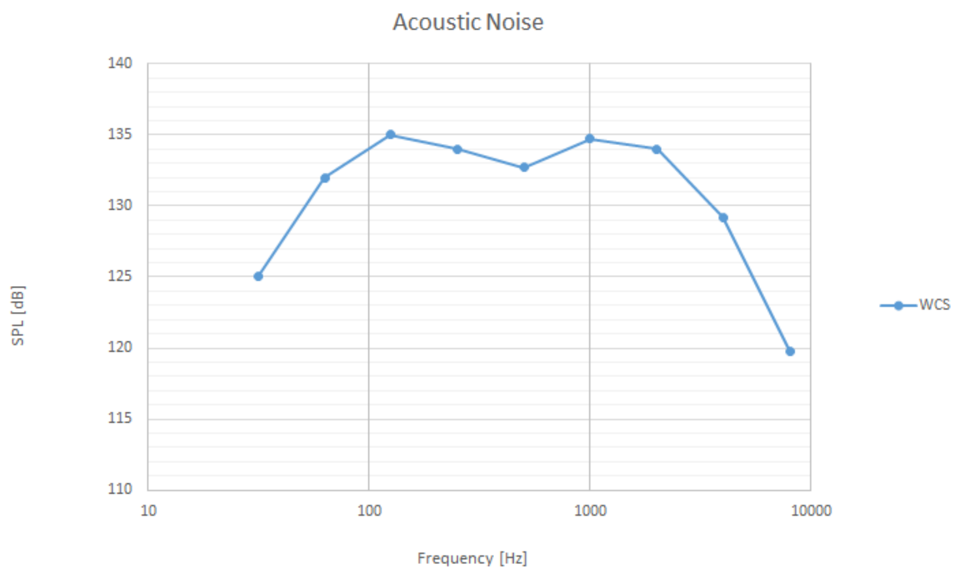

Figure 4.8: Worst Case Scenario of Acoustic noise (information in agreement with Figure 4.4).

Shock

The most demanding shock condition for MECSE, in the options studied is presented in Figure 4.9 and the shock loads values are described in Table 4.9. It is measured in the interface between the launch vehicle and the spacecraft.

Minimum allowable frequency

The minimum allowable frequencies are 45 Hz in the lateral axes and 90 Hz in the longitudinal one.

Table 4.8: Worst Case Scenario of Acoustic noise values (information in agreement with Figure 4.4).

	Frequency [Hz]	SPL [dB]
Accoustic Noise WCS - Worst Case Scenario	31.5	125
	63	132
	125	135
	250	134
	500	132.7
	1000	134.7
	2000	134
	4000	129.2
	8000	119.8

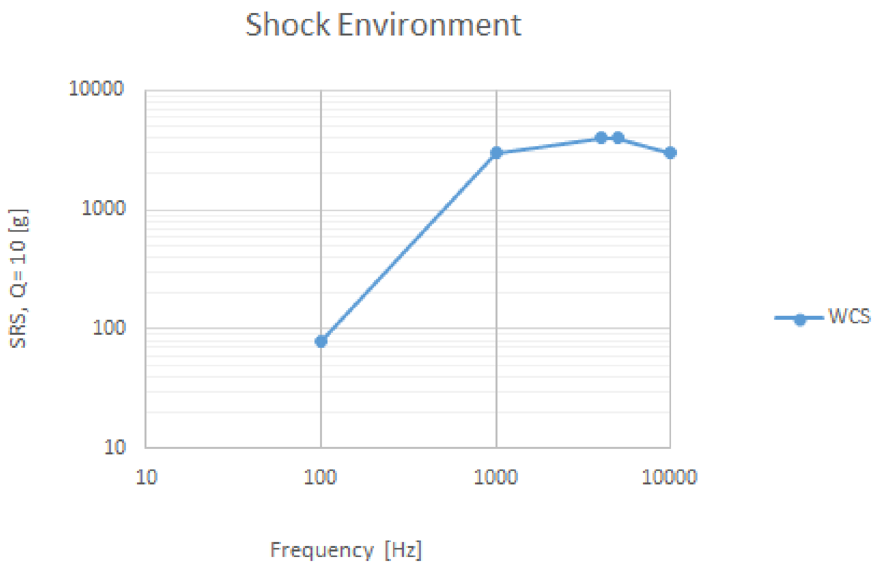

Figure 4.9: Worst Case Scenario of Shock vibration environment (information in agreement with Figure 4.5).

Table 4.9: Worst Case Scenario of Shock vibration environment values (information in agreement with Figure 4.5).

	Frequency [Hz]	SRS Q = 10 [g]
Shock environment WCS - Worst Case Scenario	100	80
	1000	3000
	4000	4000
	5000	4000
	10000	3000

4.4 Launch Vehicle Study

In order to evaluate an appropriate launcher for MECSE, Table 4.10 was created. It is a possible approach to the launch vehicle selection since, the categorization of each parameter should be more precisely defined, but it is an interesting start for a future choice of the launch vehicle for this project.

The category evaluation was made from 5 to 1, with 1 representing the most stringent and worst conditions to apply in MECSE and 5 the simplest and easiest conditions to be achieved. Rockot was not considered in this selection, due to the Eurorockot program for Rockot at the moment stand on hold, which would impair the option to launch MECSE with this rocket.

The different disciplines considered in this analysis were the launch cost, has a 25 % weight; the orbital suitability, with another 25 %; the mechanical environment, which is interconnected with the satellite mass and intrinsically with the cost to launch, with 20 %; the external mechanical interface, with 15 % relevance; the launch site, with 5 % of influence; together with the test facility and the sustainability, each one with the same importance, 5 %.

The price has been tagged according to Ref. [131]. It presents the launch cost for each kg of payload. It was used a maximum payload of 5 kg, due to the Refs. [146, 156] reports that this is the baseline payload mass for a 3U CubeSats and it is roughly the mass of a 3U CubeSat and the P-POD. The launchers with no reference or unreal data for the specific case were taken into account the standard price marked at 250k € [156].

The evaluation of the cost for launching a 3U CubeSat with 5 kg is presented with: if the cost was more than 250k € the respective level would be 1; if the price was set at around 250k €, corresponding to the standard launch price, the level 2 was attributed; if the launch cost were at the range of 250k € to 175k € it would be level 3; from 175k € to 125k € the respective level would be the 4th and less than 125000 € would be the 5th level. For Falcon 9 the standard cost is used, due to the price per kg from Ref. [131] was set at 2.7k € which would not be a realistic price for a 3U CubeSat with 5kg.

The orbital suitability was categorized evaluating, whether the launcher can deliver in the perfect orbit or into the ISS with the level 5; if there was only existed a slight change (less than 7.5º) in the inclination of the orbit, the level 4 was assigned; with a considerable change in the inclination (greater than 7.5º) and altitude the 2nd level was chosen; and completely out of the orbit altitude or inclination the level 1 was the specified.

The mechanical environment was evaluated with main attention to random vibration loads, acoustic noise and shock environment. Setting the launcher with the most restricted loads with the level 1, which indicates the heavier loading in the structure supports, and therefore an increase in mass; and the launch vehicle with the modest loads with level 5. Bloostar is categorized with level 3, due to lack of information on the launch environment of this launcher.

The mechanical interface evaluates more accurately the external interaction between CubeSat and launcher, and the launcher suitability to accommodate a CubeSat. If the launchers are not optimized for this type of launches and would be impossible to launch a 3U satellite from this launcher, it is categorized with the 1st level. The level 5 was granted if the launcher is fully prepared to launch this type of spacecraft.

The launch site category was defined in agreement with the proximity from Portugal, the type of transportation and time spent in transporting the satellite to the launch facility. The subject test facilities are added in order, to evaluate the company's willingness to provide facilities and resources to be used for qualification and acceptance tests of the model prior to its launch. The last category, the sustainability of the project evaluating the possible re-utilization of some part of the launch vehicle, being the only one with this functionality the launcher Arion 1 [138].

According to the study presented in Table 4.10, it is possible to notice 4 main solutions with higher scores, the Spanish Bloostar and the American launchers dedicated to micro and nanosatellites Electron, LauncherOne and Vector-R. It is important to point out that Bloostart integrates this list nevertheless, the mechanical environment was evaluated on the 3rd level, due to the lack of information about this topic, which may lead to an over-evaluation or more probably, in consideration with Bloostar launch profile, to a defeat evaluation. All of the launchers with an evaluation higher than 3.6 are two propulsion stages and with a payload mass accommodation under 150 kg, excepting the LauncherOne with a payload mass capacity over 300 kg. Falcon 9, Soyuz and Vega launchers with an overall classification of 3.15 are other suitable solutions, and Arion 1 and Dnerp-1 the launch vehicles with more difficulty to be adapted to the MECSE project.

Table 4.10: Launch Vehicle Suitability.

Launch Vehicle	**Cost 25%**	**Orbital Suitability 25%**	**Mechanical Environment 20%**	**Mechanical Interface 15%**	**Launch Site 5%**	**Test Facility 5%**	**Sustainability 5%**	**Overall**
Arion 1	3	2	**4**	3	**4**	4	**5**	**3.15**
Bloostar	4	4	3	**5**	**4**	2	1	**3.7**
Dnerp-1	2	4	2	2	3	2	1	**2.5**
Electron	4	4	3	**5**	2	2	1	**3.6**
Falcon 9	2	**5**	3	3	2	4	1	**3.15**
LauncherOne	**5**	4	1	**5**	**4**	**5**	1	**3.7**
Soyuz	2	**5**	**4**	2	2	3	1	**3.15**
Vector-R	**5**	4	2	**5**	2	2	1	**3.65**
Vega	2	4	3	**5**	2	3	1	**3.15**

4.5 Verification Approach proposal

A summary of the verification process and requirements are presented in Figure 4.10 with a reference to a possible verification approach to be implemented in MECSE. It indicates the different themes and conditions to be considered, such as the numerical and experimental models to build, the tests and analyses to be performed and the supporting documentation of each situation. To note that Figure 4.10 has been compiled with the information collected and presented in the previous chapters and with the MECSE project conditions. This proposed verification approach can and probably will be changed to meet the set of requirements requested by the chosen launch vehicle.

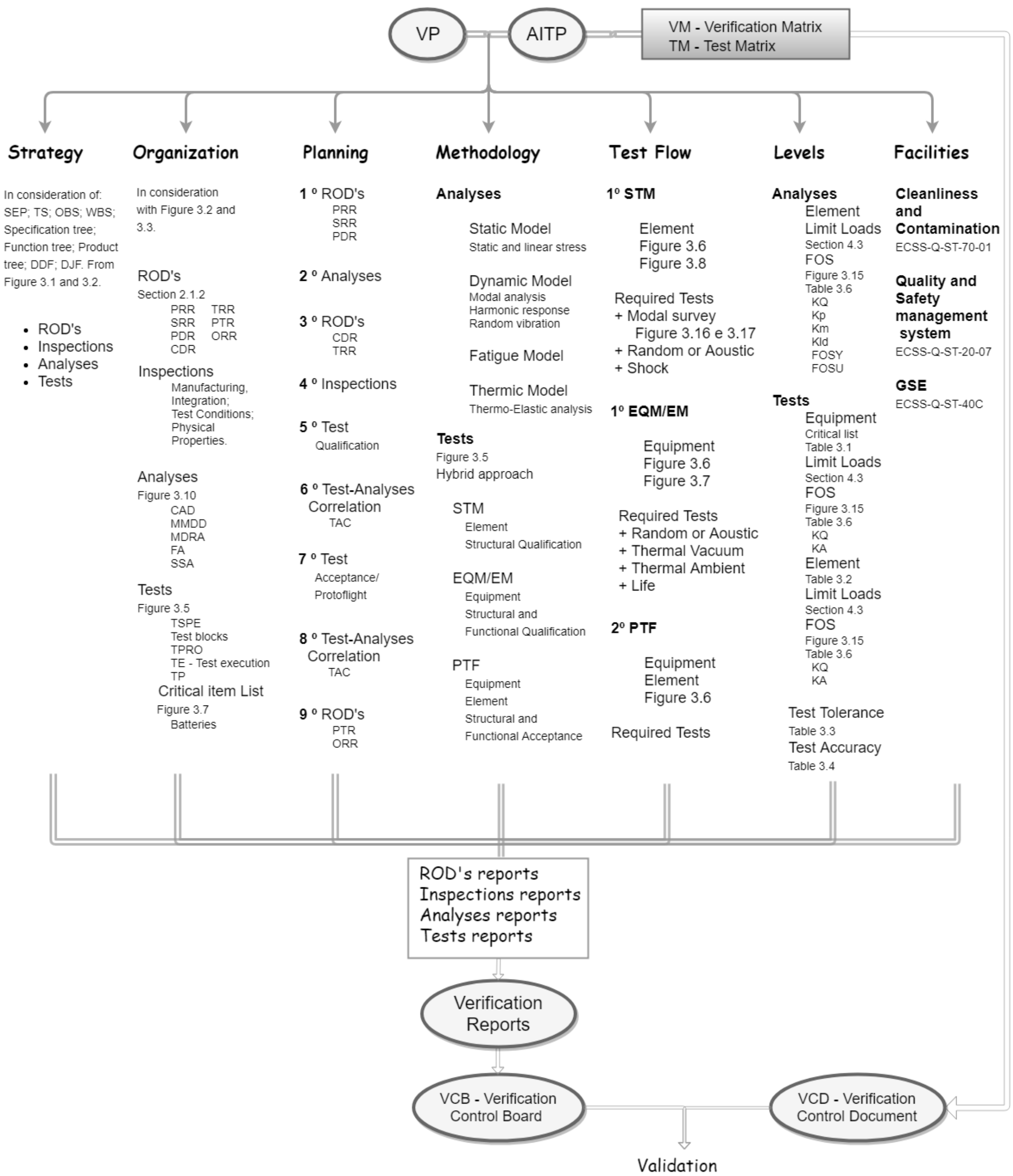

Figure 4.10: Verification Approach proposal for MECSE.

Chapter 5

Conclusion

In chapter 4 not only the launch vehicles mission requirements were presented, as well as the external interface requirements and the limit loads to apply on MECSE or in a CubeSat with similar characteristics.

In this chapter 5 the important steps and some of the key documentation to be fulfilled will be identified, when the satellite validation is the goal to achieve. To finalise this dissertation, it is presented the section 5.2 - Accomplishments, the section 5.3 - Difficulties and the section 5.4 - Future Work that can be explored in this area of study.

5.1 Important Steps

In the definition of a satellite structure, the requirements are the guidelines that must be followed in a thoughtful and coordinated way for a final product could perform the expected mission.

The requirements that every satellite should have in consideration to complete a structural configuration, design, dimensioning, verification and validation phases are the requirements presented in ECSS standards that make part of the Space engineering branch and belong to the mechanical engineering discipline, E-30 and to the system engineering discipline, E-10. Being part of these disciplines the Structural general requirements standard, ECSS-E-ST-31C; the Verifications standard, ECSS-E-ST-10-02C Rev.1; and the standard Testing, ECSS-E-ST-10-03C Rev.1, Refs. [50, 74, 82].

From the Structural general requirements standards is possible to highlight a set of the main requirements, such as the mission requirements which indicate the mechanical environment and the lifetime of the structure; from the functionality requirements the conditions to be defined are the thermal characteristics, the dynamic behaviour and the structural strength of the structure. These mission and functionality requirements are linked to the design requirements, like the factors of safety or the approved materials to use in space. Moreover, still in the structural general requirements the interface requirements indicate the available options for connecting the different satellite parts, including the interconnections between the satellite and the launch vehicle. Changing to the Verification standard, it defines, among other restrictions, the parts that must be checked for each type of verification and at what stage of the project should be performed, the structural finite element models requirements are also points to clarify, in order to evaluate the suitable conditions to create, verify and validate the numerical model. The test requirements are defined by the test objectives and by the test conditions. Finally the deliverable documents are other requirements to focus attention on, for a satellite structural definition.

The leading specifications for the definition of a structural configuration are delineated by the mission requirements along with the interface requirements. When the aim is to design a satellite structure once again, the mission requirements are a significant contribution, but in this situation when the mission requirements are related to the functionality requirements and to the design requirements. The dimensioning of the satellite components contemplates all the requirements expressed, which are the mission requirements, the functionallity requirements and the design requirements. In the verification and validation process the main inputs are the analyses, the tests and other verifications alongside with the deliverable requirements.

The verification plan is established with the definition of the critical verification conditions which are the strategy, organization and planning adopted. However, the test methodology identifies a large part of the project development, since the methodology adopted defines which are the experimental verifications to carry out and it also defines part of the product development schedule. The product schedule is significantly affected by the number of models to build, the test flow and the type of the products used.

For a complete structure validation, the documentation that is usually requested by the launch vehicle to the satellite project responsible in order to assess the structural capabilities and characteristics of the spacecraft are:

- The ICD's - Interface Control Documents explaining the Launch vehicle - spacecraft external interface points; presenting the considerations adopted in the design and configuration of the spacecraft and indicating the verification documentation, such as AIVP, the verification matrix and the test matrix, alongside with the verification reports and the verification control board report;
- The mass properties of the spacecraft and the CAD - Computer Aided Model where several characteristics of the satellite are identified, such as the centre of gravity and the inertial data;
- The finite element models to evaluate the satellite response and to assure that the coupled loads analyses are appropriate for the launch vehicle in study.

The mechanical environment that a satellite needs to overcome is defined between the ground, handling and transportation, the launch environment and the mechanical conditions that the satellite will face in orbit. The main static loads are experienced on the ground, handling and transportation with the worst case scenario of this load to have magnitudes of $\pm$ 3.5 g in the longitudinal axis, $\pm$ 2.5 g in the lateral ones and $\pm$ 6 g in the vertical ones. At the launch, the environment is characterised by the most demanding quasi-static loads, sine frequency loads, the random vibration loads, the acoustic loads and the shock loads presented in chapter 4.

In the development of this work, it was also clear that the micro and nano launchers with two propulsion stages are the launch vehicles with more favorable conditions to accommodate MECSE due to the points analysed in section 4.4. The launch vehicle with the most demanding mechanical environment is the LauncherOne that although having a bold approach to the ascent launch profile, it has the most demanding mechanical environment from the launchers analysed in this dissertation.

5.2 Accomplishments

The main aims of this work were achieved. With the definition of a set of specifications for a structural verification and validation of MECSE CubeSat.

The essential tasks and phases for the development of a product were identified, the intrinsic documentation that need to be followed and fulfilled was recognized, the definition and clarification of the different requirements for verification and validation as well, ending up with the definition of the analyses, tests and the respective limit loads for experimental and numerical verification. It was not possible to achieve the last point of this dissertation objectives since, the identification of the facilities condition to perform the various tests in the model was not evaluated.

5.3 Difficulties

In the development of this work some obstacles were found, the main difficulties were: the research on standards contemplating structural requirements from other organizations besides ESA and NASA, such it was introduced in chapter 2; and it was also a difficulty the identification of the test execution specifications, due to the lack of data on how the experimental tests should be performed, but also due to the impossibility of carrying out the required tests, since the project still is in the preliminary definition phase, which does not include performing any test.

Another important and difficult subject was the collection of data on the correlation criteria between analyses and tests, and understand how a specific numerical model can be validated efficiently. It was also challenging to comprehend the conditions and variables of the CubeSat and small launch vehicle market, due to the recent growth of these areas the different launchers companies have completely different conditions and offer different types of solution, which associated with the large sums of money required was not easy to define the different categories that should be part of the launch vehicle case study.

5.4 Future Work

For a future work and with MECSE project development in mind, it is important to perform the different numerical verifications presented in Figure 4.10. It is also mandatory to compile the documents for the definition of every test, such as the test matrix, the test specifications, procedures and test execution, also the TAC - Test-Analysis Correlation needs to be compiled. The complete fulfilment of the AIVP with information form this dissertation should also be a priority but it just would be possible when more aspects of this project are completely defined.

It would be a meaningful help if the definition of the launch vehicle that will be responsible to transport MECSE into orbit is performed, in order to achieve a precise definition of the launch conditions and the external interfaces requirements, which will lead to a more efficient dimensionality of the structural components. The selection of the launch vehicle would also help in the discussion of requirements, conditions, numerical verifications, experimental verifications and documents that would be requested for MECSE validation, and it would also allow defining more specifications in the experimental verifications, for example if the notching technique is a possible option for these project or it is not.

5.5 Conference

The work developed in this dissertation regarding the structural requirements and the case study on MECSE has already been presented in the Workshop and Advanced School with the title: "11º International Spacelight Dynamics and Control" that took place on September 26 until 28, in Covilhã, UBI. It was organized by the C-MAST and by the SpaceWay [157].

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Appendix A

Nanosatellites and CubeSat Database

In Appendix A is presented the evolution of nanosatellites and CubeSats market over the past years.

Figure A.1 was collected from Ref. [35], and it represents the growth of CubeSats and mainly the growth of the CubeSats with commercial aims. The last data update of this figure was made from Spaceworks, NSR - Northern Sky Researcher among other databases and the reshearch about this topic was made at 21/03/2018.

Figure A.2 represents the evolution of nanosatelites from 1998, until the 2023 prediction. In Figure A.3 is featured the number of CubeSats built and launched since the standardization of a CubeSat.

Figure 1.2, Figure A.2 and Figure A.3 were collected from Ref. [24] which had the last data update at 01/01/2018 and the reshearch on this topic was made at 21/03/2018.

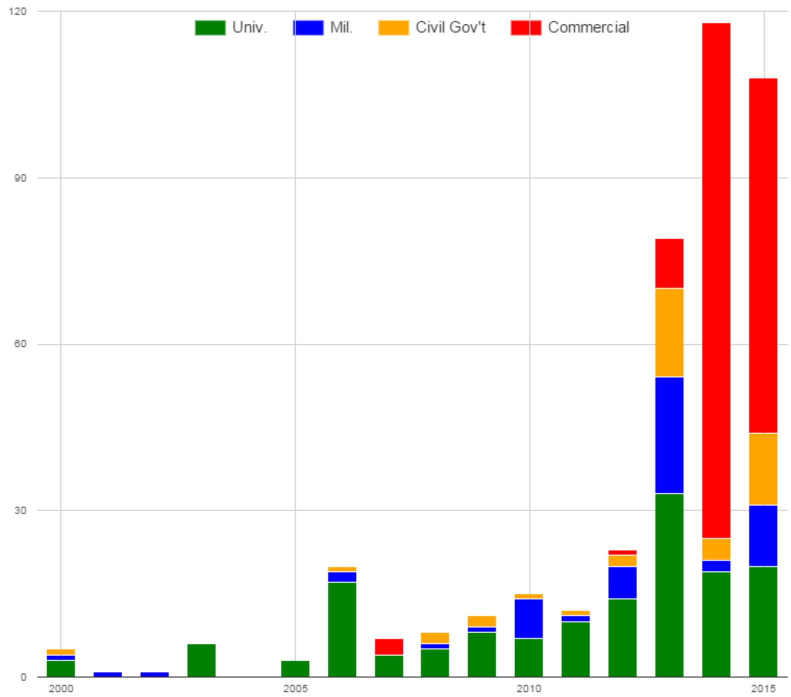

Figure A.1: CubeSat mission category from 2000 until 2015 (data from Ref. [35]).

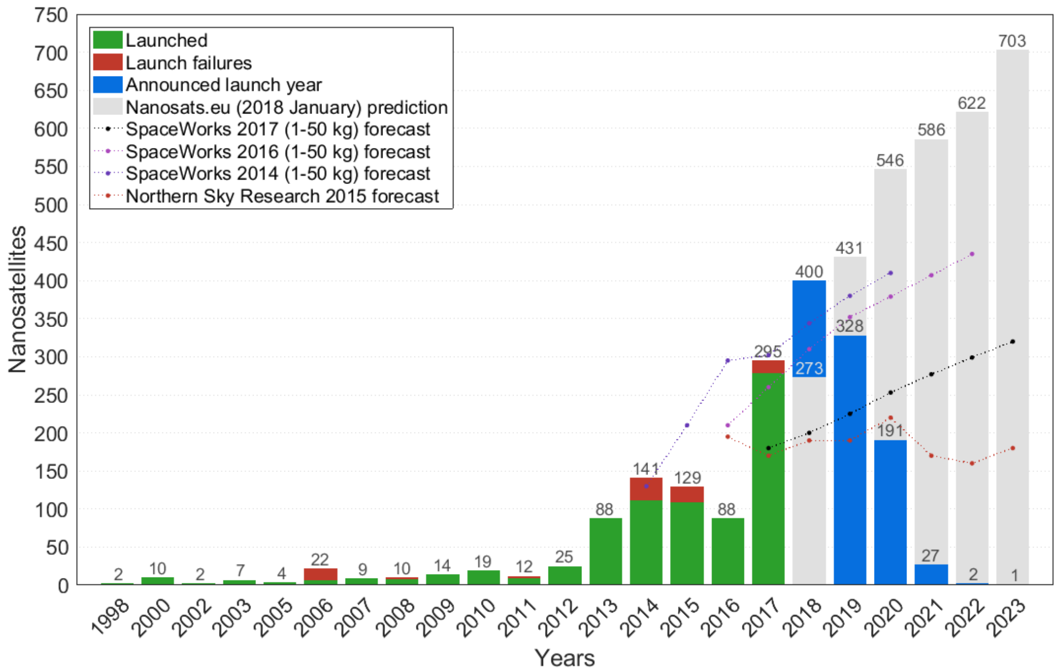

Figure A.2: Nanosatellites launched by year, evolution from 1998 to 2023 (predictions from Ref. [24]).

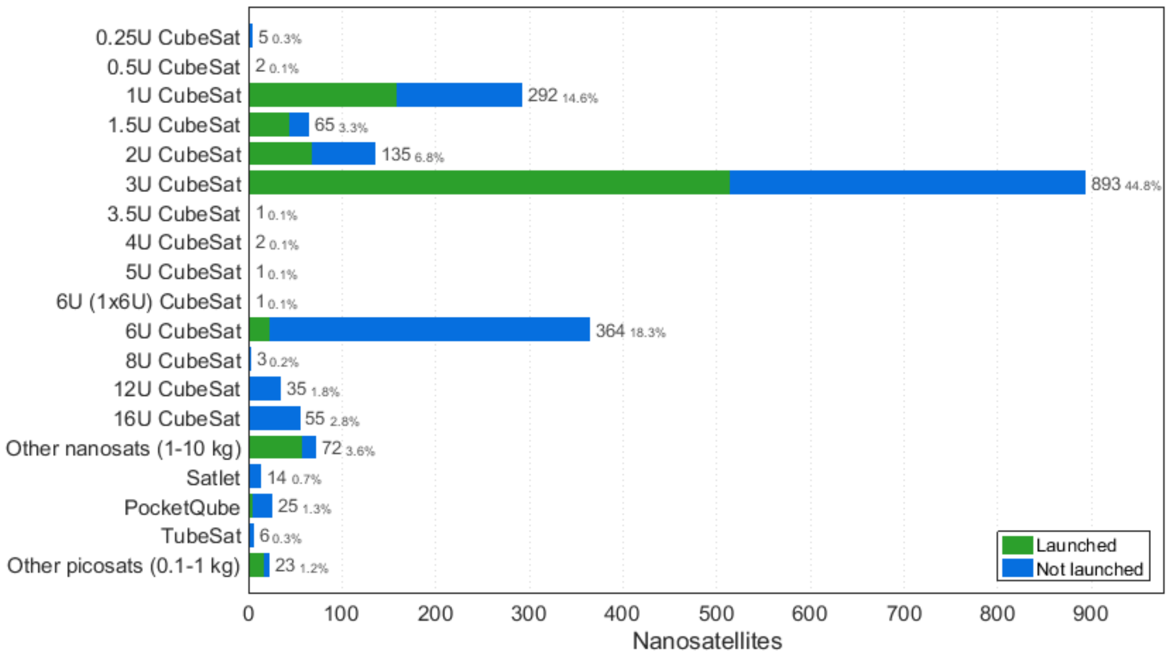

Figure A.3: Nanosatellites by type (figure from Ref. [24]).

Appendix B

MECSE Specifications

B.1 MECSE Orbit

This section B.1 represents the main orbit conditions that MECSE should perform.

Table B.1: Orbital details of MECSE (information collected from Ref. [44]).

Epoch	1-Jan-2020
Orbit Type	LEO
Altitude of Apogee	350 km
Eccentricity	0
Inclination	52,6º
Argument of Perigee	0º
RAAN	0º
True Anomaly	0º
Orbital Period	1,52 h
Orbital Velocity	7,7 km/seg

B.2 MECSE Structure

In Figure B.1 is identified pertinent points of MECSE with the representation of each part and element that MECSE is intent to comtemplate.

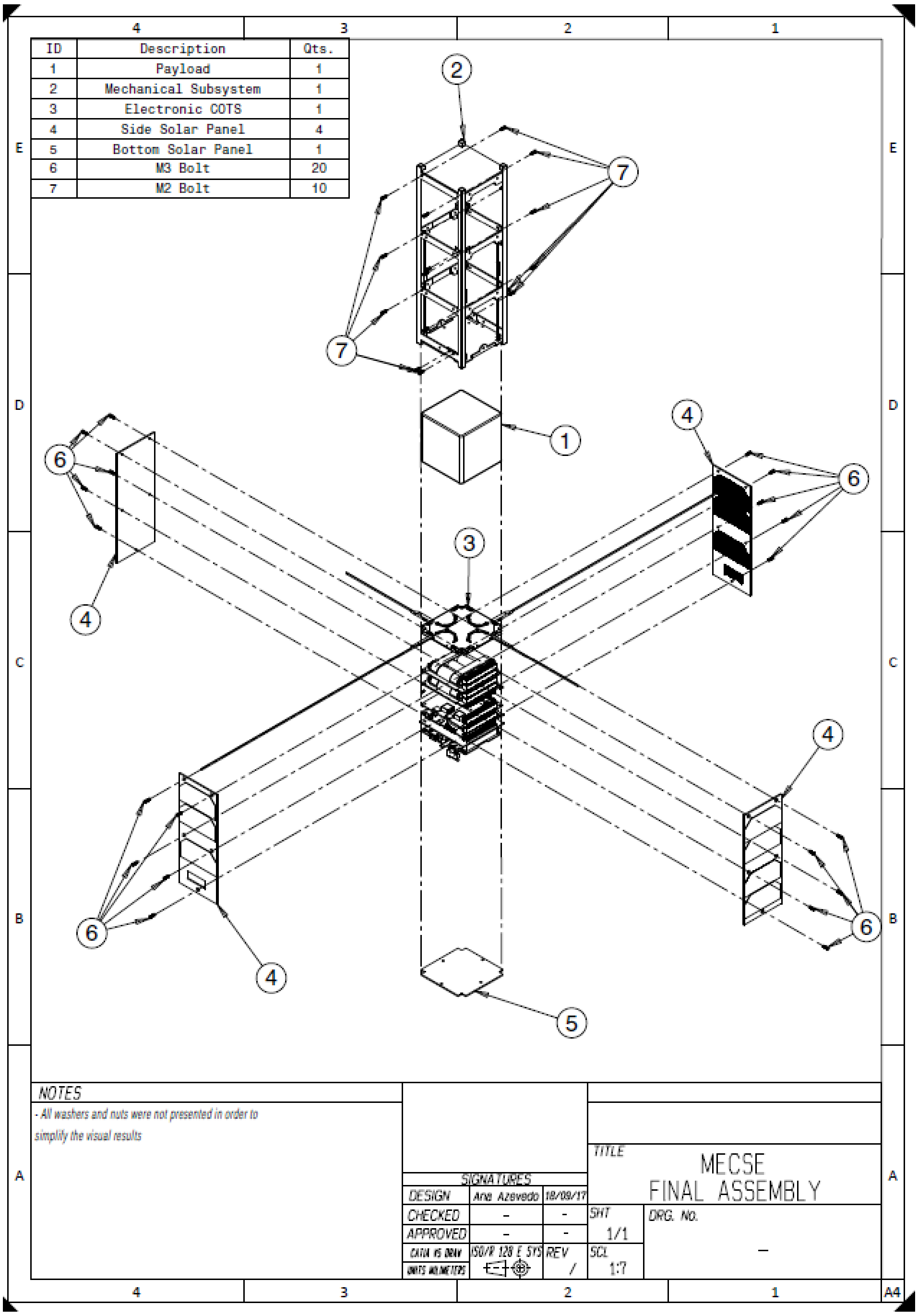

Figure B.1: MECSE's Assembly view (data collected from Ref. [47]).

Appendix C

System Engineering

Figure C.1 shows the system engineering sub-functions, their inter-relationships and their main activities during the system engineering process.

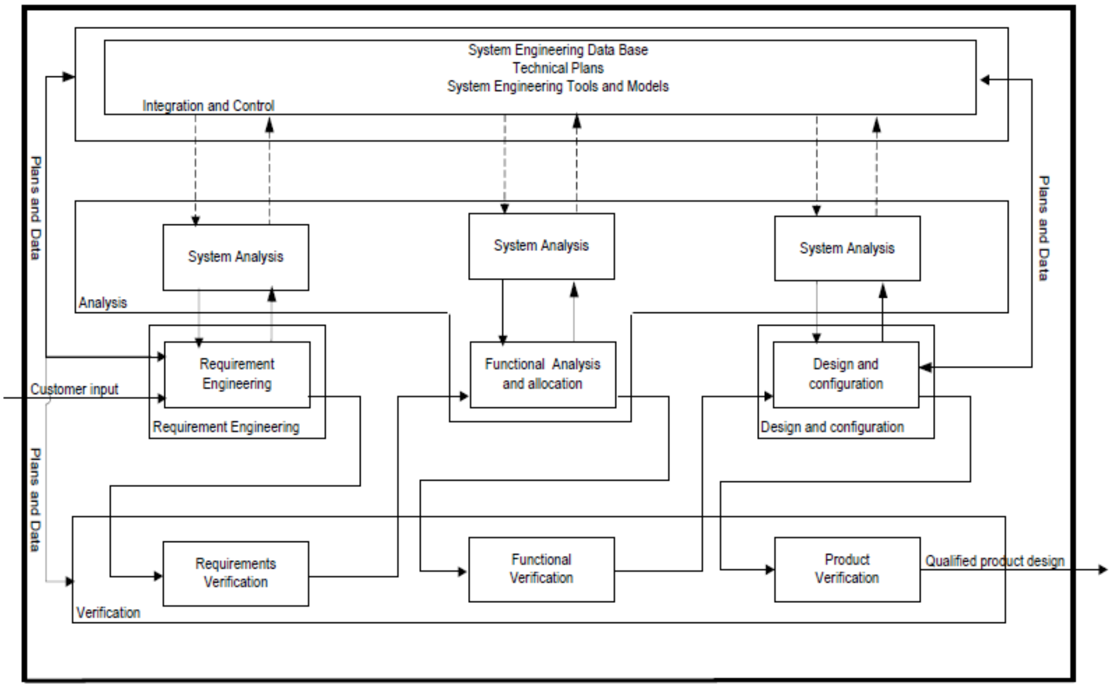

Figure C.1: System Engineering inter-relationships (figure from Ref. [62]).

C.1 Project Phasing

ESA

The documentation and reviews that should be provided, in order to the next phase could proced are identified below identified by each phase. The set of documentation and reviews will be explicit in agreement with Refs. [61,62] and it has the purpose to explain the importance of each document.

Phase A - Feasibility

- PAP - Product Assurance Plan describes the activities to assure the quality of a space product in agreement with the specified mission objectives and with the aim of demonstrate compliance of the procuct assurance requirements [69].

- VP - Verification Plan is the document that defines the verification strategies, approaches, the model philosophy, the product matrix, the tests, inspections, analyses, verification tools and all the involved documentation. The control methodology and the test plan are others responsibilities for the management and organization verification defined by the VP [50].

- VCD - Verification Control Document is the document responsable for part of the qualification and it includes the Verification Matrix. It has in account a list of requirements to be verified according to the level of the product, the stage and the method to perform the verification [50].

- The review to perform on this phase is the PRR - Preliminary Requirements Review and it has the objectives of releasing the preliminary management, engineering and product assurance plans, the technical requirements specification whit the confirmation of the technical and programmatic feasibility [70].

Phase B - Preliminary Definition

- OBS - Organizational Breakdown Structure, is the document that indicates the key personnel and the assigned responsibility parts for each work stage [70].

- The review to perform on this phase is the SRR - System Requirements Review, it has the goals of releasing the updated TS and validate the preliminary design definition and verification program [70] and also;

- The other review is the PDR - Preliminary Design Review, it has the duty to confirm the preliminary design, basically verify the selected concept and the technical solutions against the project and system requirements; release the final PMP, SEP and PAP; deliver the product tree, the WBS, the specification tree and the verification plan [70].

Phase C - Detailed Definition

- AITP - Assembly, Integration and Test Plan, or just Test Plan is the master plan to produce the AIT - Assembly, Integration and Test process and it describes the AIT activities and verifications, from tools (facilities and GSE - Ground Support Equipment), documentation, management, schedule, among other characteristics [74].

- The review to perform on this phase is the CDR - Critical Design Review and it has the objectives to evaluate the qualification and validation status of critical processes or elements; confirm the compatibility with external interfaces; release the final design, the flight hardware/software, the AITP, as well has the responsibility for delivering the user manual [70].

Phase D - Qualification and Production

- The review to perform on this phase is the QR - Qualification Review. It is done to confirm that the the design meets the requirements, and to verify the acceptability of all waivers and deviations [70].

- The other review is the AR - Acceptance Review, it has the duty to confirm all the verification process and with that demonstrate that the product is free of workmanship errors and is ready for subsequent operational use. This review checks if the acceptance verification record is complete at all levels, it verifies if the product is ”as-built” and if its constituent components are consistent ”as-designed”. It also verifies the acceptability of all waivers and deviations. Authorize the deliver, and release the certificate of acceptance is the ultimate goal for the AR, in Phase D [70].

- This phase still has another review the ORR - Operational Readiness Review, it has the goals of demonstrate the readiness of the operational procedures and their compatibility with the flight system and to the operations team, alongside with the acceptance and releasing of the ground segment for operations [70].

Still exist other important documentation such as the:

- ICD’s - Interface Control Document’s which define the interfaces between all the subsystems of a space vehicle [60].

- and the SOW - Statement Of Work which defines the allocation of specific engineering requeriments per phase depending on the business agreement [62].

C.1.1 Documents per Delivery

This subsection (constituded by Table C.1, Table C.2 and Table C.3) presents the documents, which support the project reviews, associated with the engineering activities, as specified in Ref. [70].

To note that, some of the tables in below, could present misguided indications about which is the ECSS document where the desired document is defined and explained. This happens due to the continued updating of the various standards.

For example, the Verification Plan, presented in Table C.2 that is explicit in ECSS-E-ST-10-02 - Annex H; actually is made clear at ECSS-E-ST-10-02C Rev.1 - Annex A [50].

Table C.1: Deliver Documents per Review (table from Ref. [61]).

Document title	ECSS document	DRD ref.	Phase 0	Phase A	Phase B		Phase C	Phase D			Phase E				Phase F
			MDR	PRR	SRR	PDR	CDR	QR	AR	ORR	FRR	LRR	CRR	ELR	MCR
Design definition file	ECSS-E-ST-10	Annex G		+	+	+	+	+							
Function tree	ECSS-E-ST-10	Annex H		+	+	+									
Product tree	ECSS-M-ST-10	Annex B		+	+	+									
Specification tree	ECSS-E-ST-10	Annex J			+	+									
Technical budget	ECSS-E-ST-10	Annex I		+	+	+	+	+	+						
Preliminary technical requirements specifications for next lower level	ECSS-E-ST-10-06			+	+										
Technical requirements specifications for next lower level	ECSS-E-ST-10-06				+	+									
Design definition file for next lower level						+	+	+	+						
Interface control document	ECSS-E-ST-10-24	Annex A			+	+	+	+	+	+	+	+			
Product User manual / User Manual	ECSS-E-ST-10	Annex P					+	+	+	+	+	+	+	+	+
Design justification file	ECSS-E-ST-10	Annex K		+	+	+	+	+							
Requirements traceability matrix w.r.t. next lower level	ECSS-E-ST-10	Annex N		+	+	+									
Requirement justification file	ECSS-E-ST-10	Annex O	+	+	+	+									
System concept report	ECSS-E-ST-10	Annex C	+	+											
Trade off reports	ECSS-E-ST-10	Annex L	+	+	+	+	+								

Table C.2: Deliver Documents per Review, partII (table from Ref. [61]).

Document title	ECSS document	DRD ref.	Phase 0	Phase A	Phase B		Phase C	Phase D			Phase E				Phase F
			MDR	PRR	SRR	PDR	CDR	QR	AR	ORR	FRR	LRR	CRR	ELR	MCR
Mission description document	ECSS-E-ST-10	Annex B	+	+											
Specifications															
Preliminary technical requirements specification	ECSS-E-ST-10-06	Annex A	+	+											
Technical requirements specification	ECSS-E-ST-10-06	Annex A			+										
Interface requirements document	ECSS-E-ST-10	Annex M		+	+	+									
System engineering plan	ECSS-E-ST-10	Annex D	+	+	+	+	+	+	+						
Technology plan	ECSS-E-ST-10	Annex E		+	+	+									
Technology matrix	ECSS-E-ST-10	Annex F		+	+	+									
Verification plan	ECSS-E-ST-10-02	Annex B		+	+	+	+	+	+						
AIT QM/FM plan	ECSS-E-ST-10-03	Annex A				+	+	+	+						
Orbital debris mitigation plan	ISO 24113		+	+	+	+	+	+	+	+	+	+		+	+
Other related plans (as called in ECSS-E-ST-10 Annex D)				+	+	+	+	+	+						
Coordinate system document	ECSS-E-ST-10-09	Annex A		+	+	+	+	+							

Table C.3: Deliver Documents per Review, partIII (table from Ref. [61]).

Document title	ECSS document	DRD ref.	Phase 0	Phase A	Phase B		Phase C	Phase D			Phase E					Phase F
			MDR	PRR	SRR	PDR	CDR	QR	AR	ORR	FRR	LRR	CRR	ELR	MCR
Verification control document	ECSS-E-ST-10-02	Annex C		+(1)	+(1)	+(1)	+	+	+	+	+	+	+	+	+
Test specification	ECSS-E-ST-10-03	Annex D					+	+	+	+	+	+	+	+	+
Analysis report	ECSS-E-ST-10	Annex Q		+	+	+	+	+	+	+	+	+	+	+	+
Mathematical model description					+	+	+	+							
Correlation report							+	+							
Test procedure	ECSS-E-ST-10-03	Annex C					+	+	+	+	+				
Test report	ECSS-E-ST-10-02	Annex D					+	+	+	+	+	+	+	+	+
Verification report	ECSS-E-ST-10-02	Annex H					+	+	+	+	+	+	+	+	+
Design justification file for next lower level							+	+	+						
Review of design report	ECSS-E-ST-10-02	Annex F					+	+							
Inspection report	ECSS-E-ST-10-02	Annex G					+	+	+						
GSE specifications						+	+	+	+						
GSE Data packages							+	+	+						
Note (1) : Document limited to the verification matrix															

Appendix D

Norms and Standards

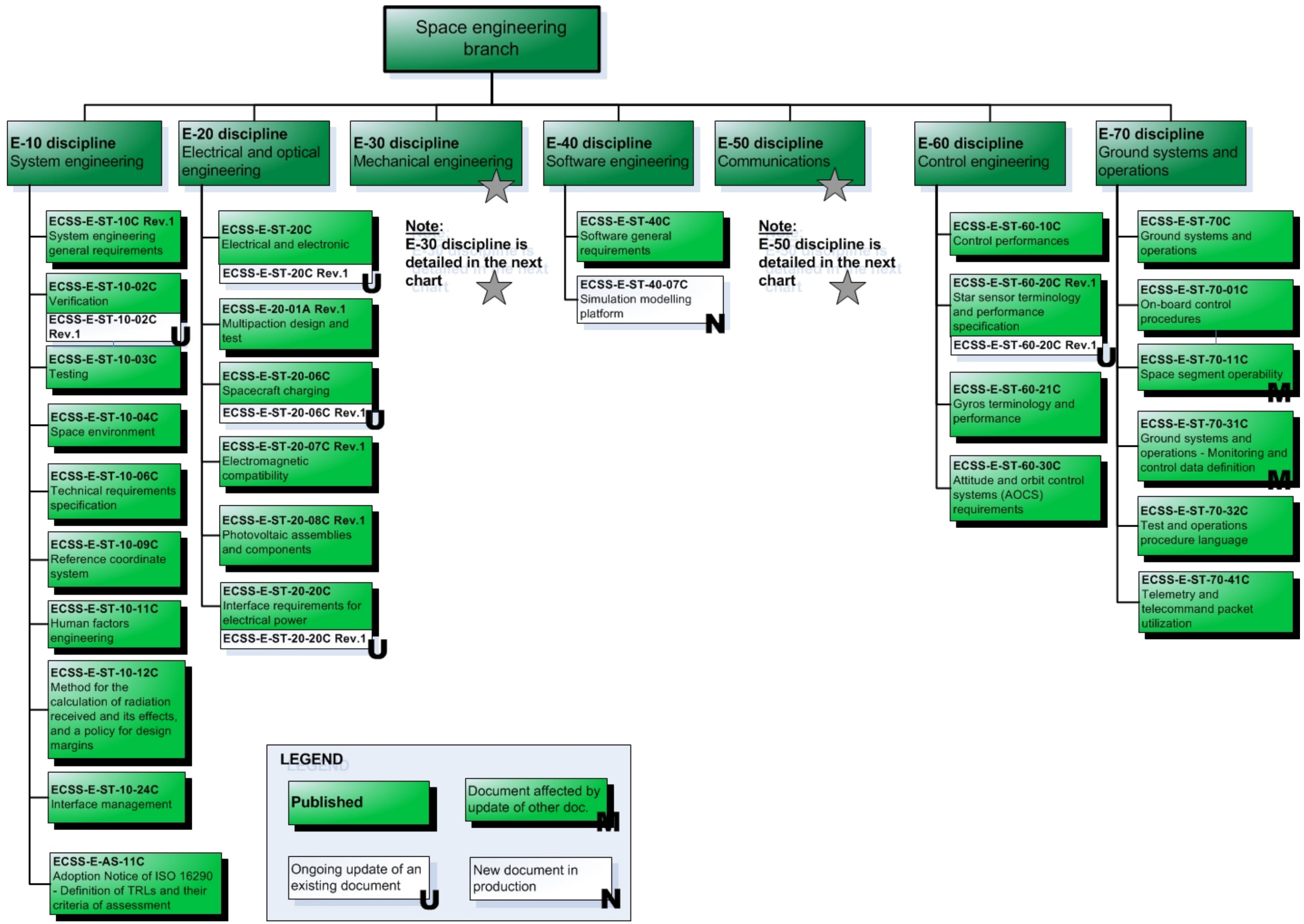

Figure D.1: ECSS Space Engineering Branch Standards (figure from Ref. [158]).

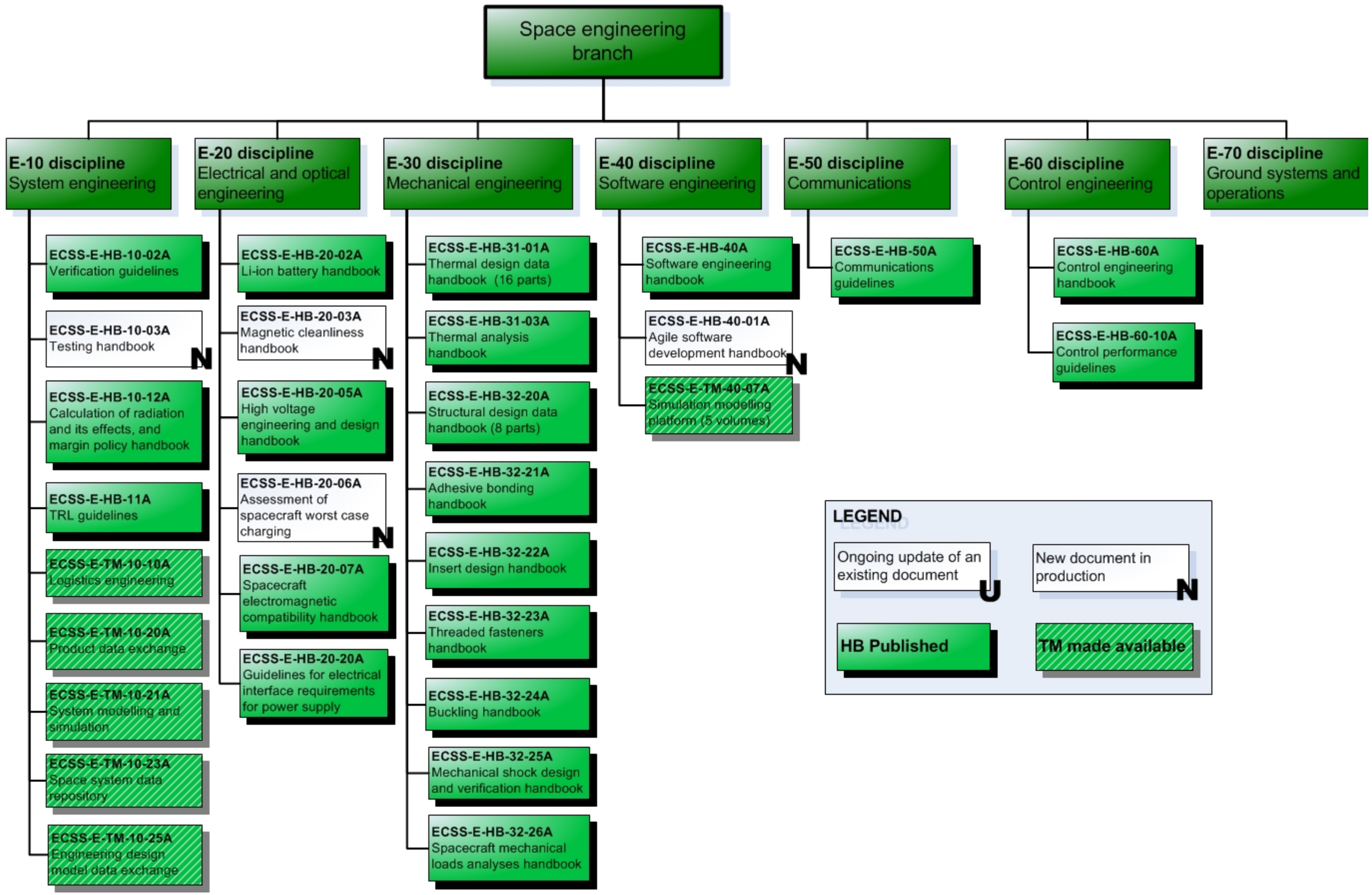

Figure D.2: ECSS Space Engineering Branch Handbooks and Technical Memoranda Branch (figure from Ref. [158]).

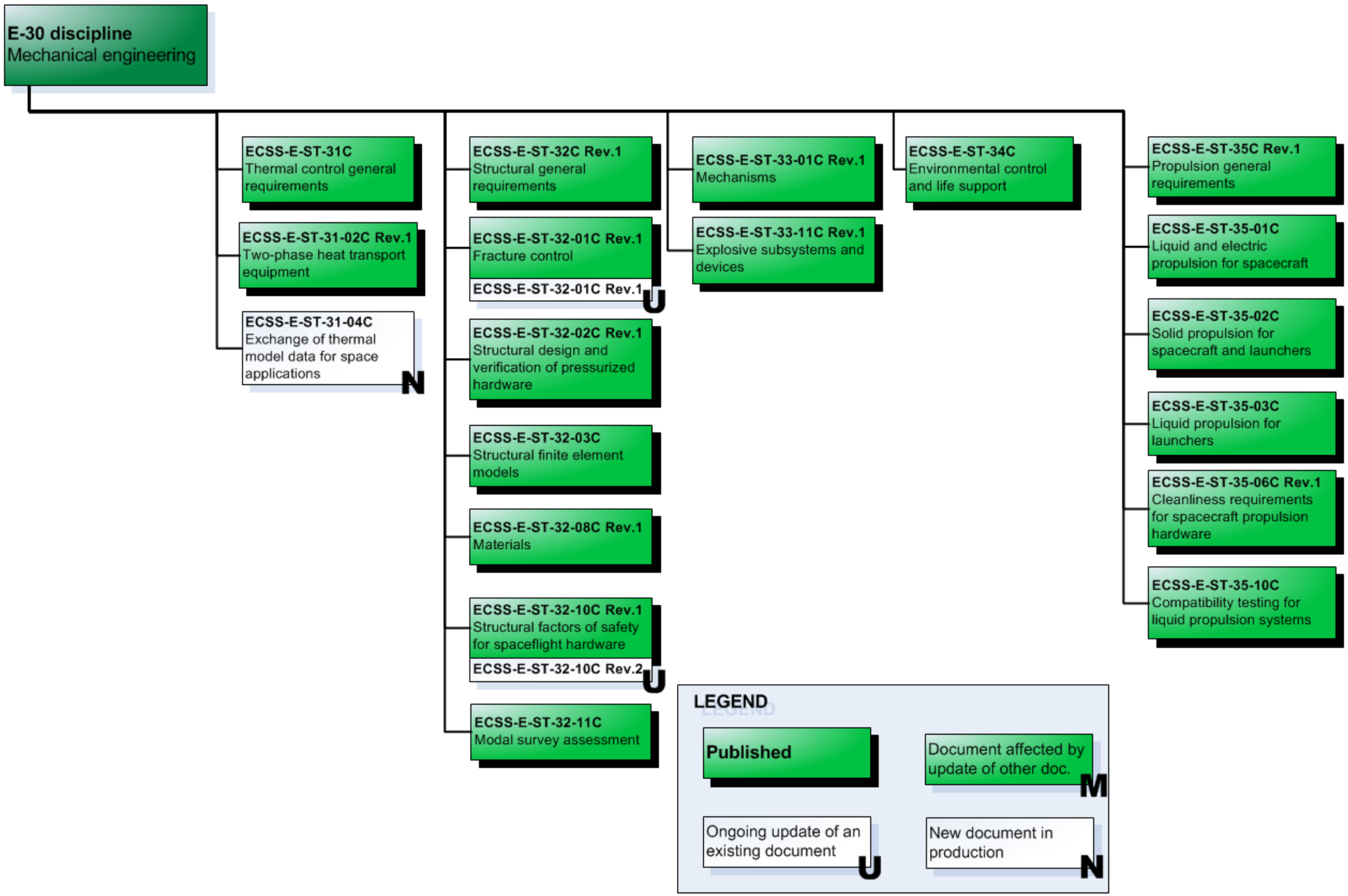

Figure D.3: ECSS Mechanical Engineering Standards tree, E-30 (figure from Ref. [159]).

D.1 Verification Plan

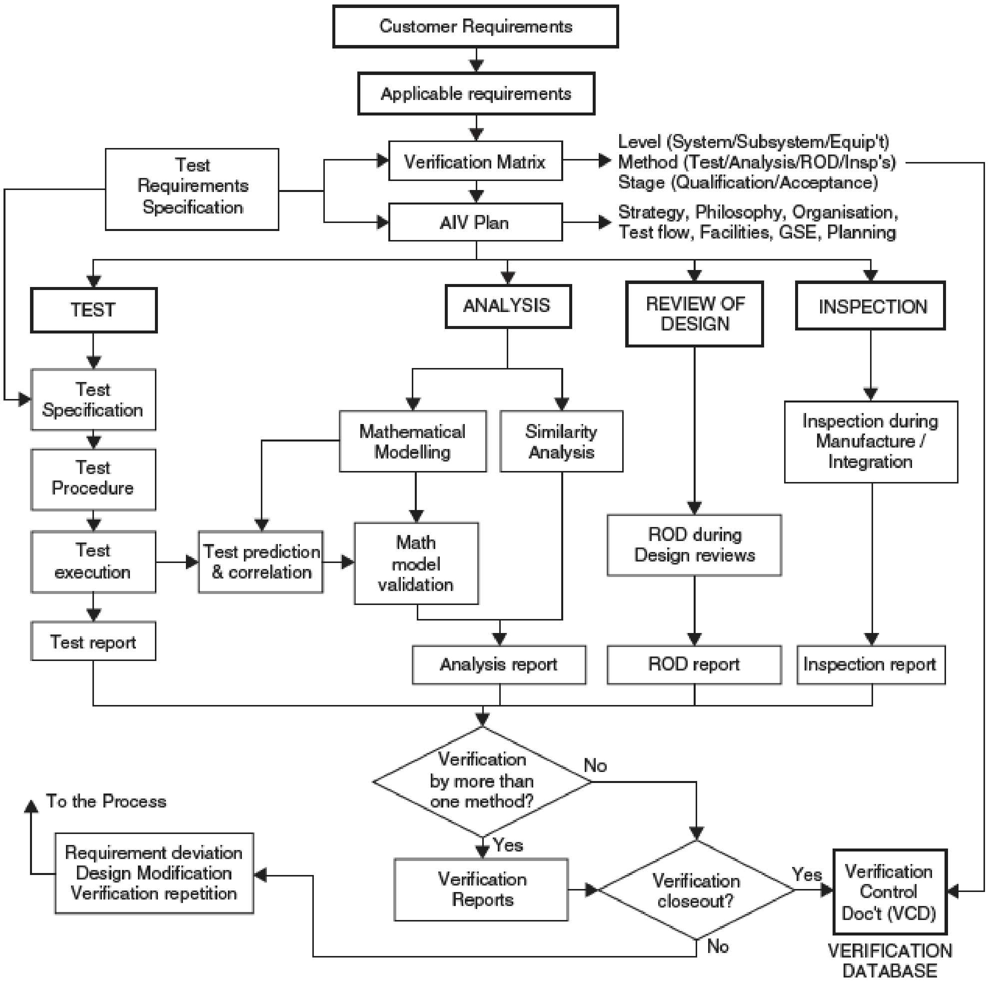

Figure D.4: Verification Planning Logic (figure from Ref. [108], representing the Figure 6-1: Verification documentation from Ref. [75]).

D.2 Systems general Requirements

D.3 Structural General requirements

D.3.1 Materials

The Figure D.5 presented below, was compiled with information referent to the standard Materials, ECSS-E-ST-32-08C.

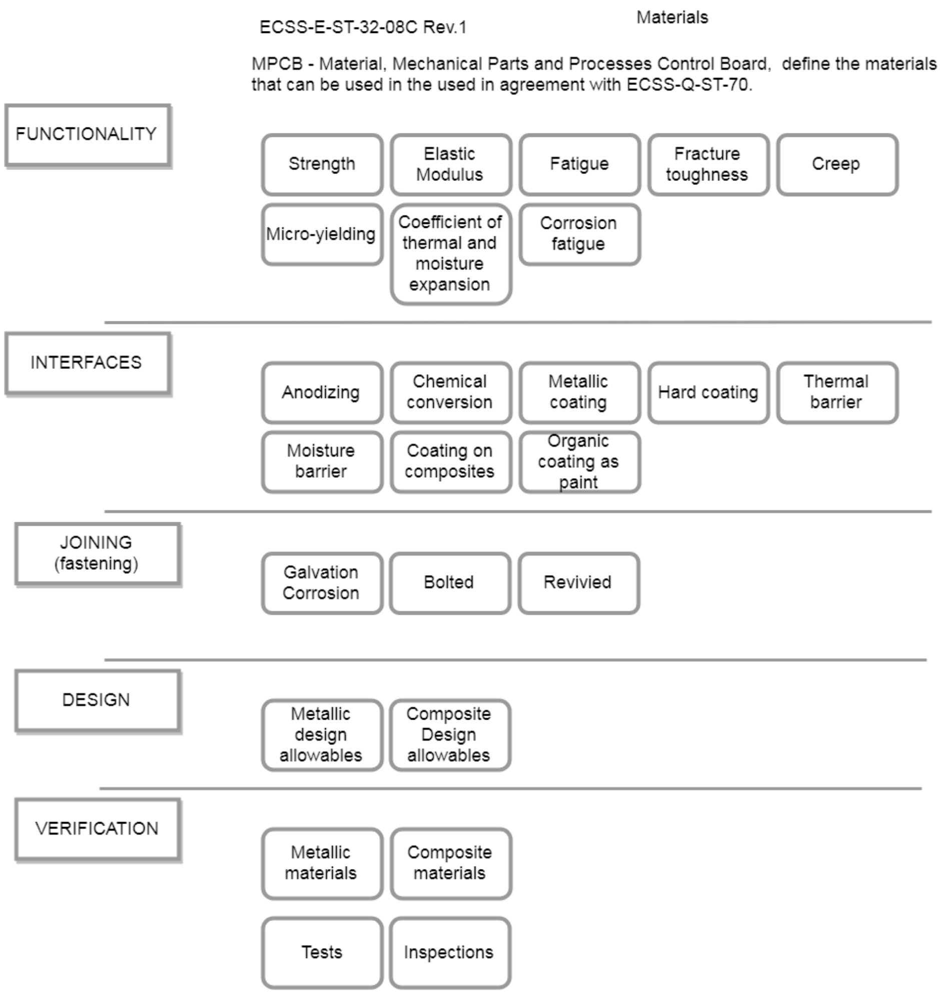

Figure D.5: Material requirements (figure composed in agreement with Ref. [88]).

Appendix E

Mechanical Environment

E.1 Launch Environment

Ref. [98], just like Refs. [43,99] report the occurrence of loads on the launch vehicle and the reasons for the generation of these loads.

Being the different phases of launching the main reason for the creation of these loads as reported in below.

Table E.1: Sources of launch vehicle environment loads, summary [98].

	Acoustics	Random Vibration	Sine Vibration	Shock
Lift-off	X	X		
Aerodynamics /Buffet	X	X		
Separation (stage, fairing, spacecraft)				X
Motor burn /Combustion/ POGO		X	X	

Steady state and low frequency loads like can be seen in the presented Table E.1 and in Figure E.1, the main sources of this load come from the stages of burn out, but also the crosswind and the manoeuvres have influence in the steady state and low frequency loads.

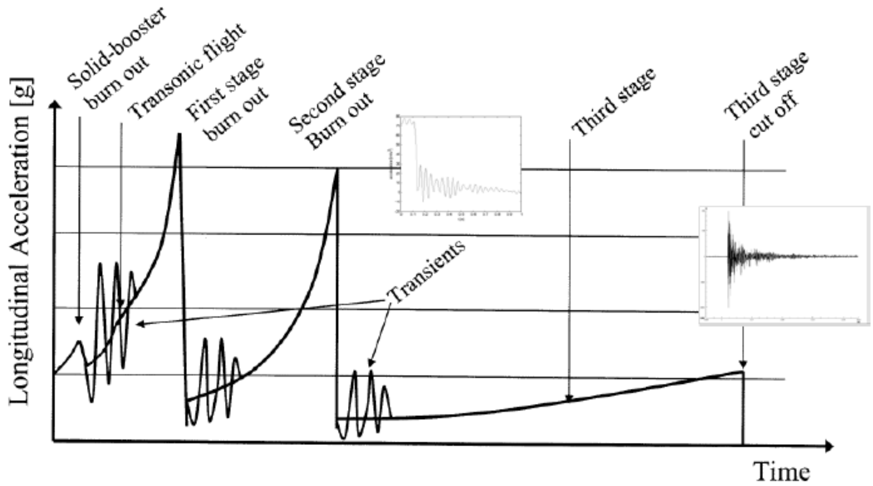

Figure E.1: Steady state and low frequency sources of loads (figure from Ref. [99]).

Random Vibration is consequence of the engines functioning, the structural response to broad-band acoustic noise and the break of the aerodynamic turbulence boundary layer. The normal distribution of this load is presented as in Figure E.2 [43].

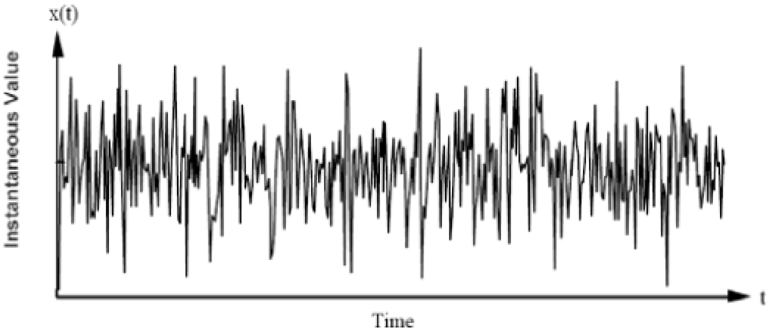

Figure E.2: Exemple of a Random signal (figure from Ref. [99]).

Acoustic loads happens mainly due to the engine functioning and to the aerodynamic turbulence because of the airflow separation along the launch vehicle. The acoustic noise impinges high responses on light weight units (solar panels are examples of it), a representation of this load can be consult in Figure E.3 [43].

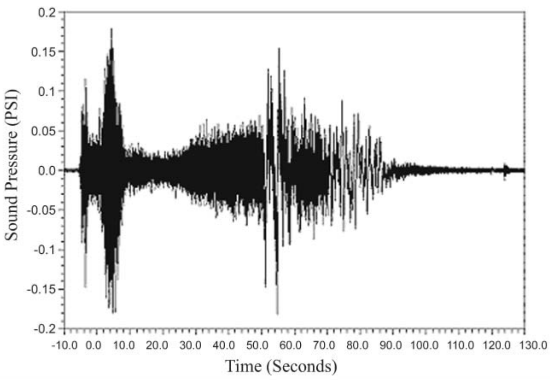

Figure E.3: Measured acoustic loads, high frequency sound pressure in function of time (figure from Ref. [43]).

Shock loads are formed due to the separation of the launch vehicle in the different launch stages. The typical acceleration history, in a separation stage, is presented in Figure E.4 [43].

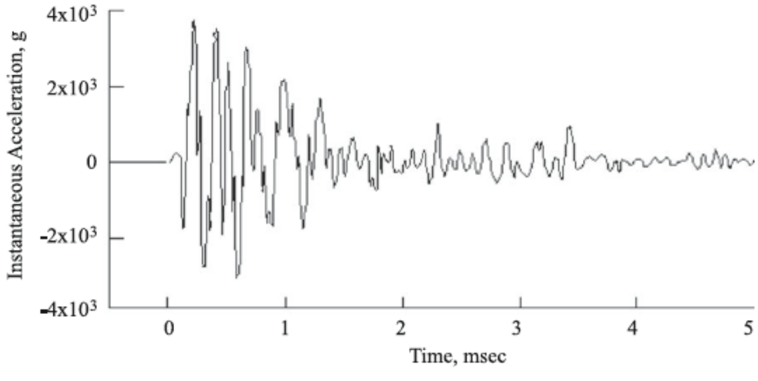

Figure E.4: Tipycal accelaration history in a separation stage (figure from Ref. [43]).